AD735766

USAAMRDL TECHNICAL REPORT 71-55

INVESTIGATION OF THE STABILATOR ON
THE S-67 AIRCRAFT

By

T. T. Kaplita

October 1971
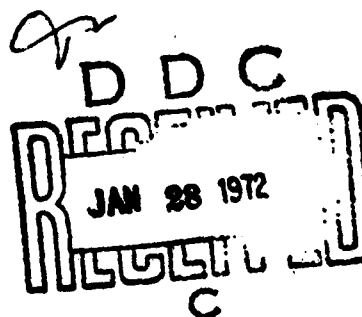
EUSTIS DIRECTORATE
U. S. ARMY AIR MOBILITY RESEARCH AND DEVELOPMENT LABORATORY
FORT EUSTIS, VIRGINIA

Approved for public release;
distribution unlimited.

D D C

JAN 28 1972

C

DOCUMENT CONTROL DATA - R & D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY (Corporate author)	2a. REPORT SECURITY CLASSIFICATION
Sikorsky Aircraft Division United Aircraft Corporation Stratford, Connecticut	Unclassified 2b. GROUP

3. REPORT TITLE

INVESTIGATION OF THE STABILATOR ON THE S-67 AIRCRAFT

4. DESCRIPTIVE NOTES *(Type of report and inclusive dates)*

Final Report

5. AUTHOR(S) *(First name, middle initial, last name)*

Thaddeus T. Kaplita

6. REPORT DATE	7a. TOTAL NO. OF PAGES	7b. NO. OF REFS
October 1971	111	7

8a. CONTRACT OR GRANT NO. DAAJ02-71-C-0010	9a. ORIGINATOR'S REPORT NUMBER(S)
b. PROJECT NO.	USAAMRDL Technical Report 71-55
c. Task 1F163204D15704	9b. OTHER REPORT NO(S) *(Any other numbers that may be assigned this report)* SER-67006
d.	

10. DISTRIBUTION STATEMENT

Approved for public release; distribution unlimited.

11. SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY
	U.S. Army Air Mobility Research and Development Laboratory Eustis Directorate Fort Eustis, Virginia

13. ABSTRACT A flight test and simulation investigation of the S-67 stabilator was conducted. The S-67 stabilator, an all-movable, in-flight-trimmable horizontal stabilizer, is coupled to the longitudinal cyclic control in forward flight and may be uncoupled to free-float in hover. The flight tests included hover, transitions, low- and high-speed flight, autorotation, and stabilator bias actuator hardovers. In the simulation study, stabilator design was varied to establish trends due to design changes.

During hovering flight, the stabilator in the "fly" (horizontal) mode did not produce the pitching oscillations experienced by the S-61F (NH-3A) helicopter. The stabilator is located so that no adverse disturbances were experienced in hover in either the "fly" or "free" mode. The "free" mode of the stabilator does not offer enough improvement in handling qualities, for the loading condition tested, to justify the control system complexity.

The existing stabilator design will provide the aircraft with acceptable trim and static stability for the full G.W. and C.G. envelope. In-flight trimmability provides control of fuselage attitude independent of the rotor. However, one trim bias angle of +2.5 degrees (leading edge up) is optimum for all loading conditions tested with the aircraft in a clean configuration. Bias angle has no significant effect on autorotation. On the S-67, in-flight trimmability provides limited benefits in high- or low-speed flight. Adequate delay times and control power are available to recover from bias actuator hardovers. Rotor control loads, however, will exceed endurance limits with hardovers above 160 knots.

DD FORM **1473** REPLACES DD FORM 1473, 1 JAN 64, WHICH IS OBSOLETE FOR ARMY USE.

DISCLAIMERS

14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Helicopter						
Winged Helicopter						
Sikorsky S-67						
Aerodynamics						
Stability and Control						
Handling Qualities						
Performance						
Horizontal Stabilizer						
Stabilator						

This report was prepared by United Aircraft Corporation, Sikorsky Aircraft Division, under Contract DAAJ02-71-C-0010.

The program was a flight investigation of an all-movable, in-flight trimmable, horizontal tail surface called a stabilator as installed on the S-67 winged helicopter. A computer simulation study was included to determine the effect of stabilator design changes on aircraft stability and control characteristics. This program is one of four flight investigations conducted on the S-67 winged helicopter. The other three flight investigations were concerned with speed brakes, a force-feel control system, and overall aircraft maneuverability. These programs will be presented in separate reports.

Due to its location outside of the main rotor downwash, the stabilator did not cause the pitching oscillations that were experienced by other Sikorsky helicopters in hover and low-speed flight. Uncoupling the stabilator from the control system does not offer sufficient advantages in hover and low-speed flight to justify the additional complexity of the control system. Although the in-flight trimmability of the stabilator provides control of the fuselage attitude independent of the rotor, one stabilator incidence angle, 2.5 degrees leading edge up, is optimum for all loading conditions tested. Because of the limited gains in both the low- and high-speed flight regimes and the increased complexity, the in-flight movable feature of the configuration is not recommended.

The report was reviewed by this Directorate and is technically correct.

This program was conducted under the technical management of Mr. R.C. Dumond of the Applied Aeronautics Division.

INVESTIGATION OF THE STABILATOR ON
THE S-67 AIRCRAFT

SER-67006

by

T. T. Kaplita

Prepared by

United Aircraft Corporation
Sikorsky Aircraft Division
Stratford, Connecticut

for

EUSTIS DIRECTORATE
U. S. ARMY AIR MOBILITY RESEARCH AND DEVELOPMENT LABORATORY
FORT EUSTIS, VIRGINIA

ABSTRACT

A research flight test and computer simulation investigation of the S-67 stabilator was conducted. The S-67 stabilator is an all-movable, in-flight-trimmable horizontal stabilizer. It is coupled to the main rotor longitudinal cyclic control in forward flight and may be uncoupled to free-float in hover. This investigation was conducted to determine if these features offer improved performance and handling qualities.

The flight tests included hover, hovering turns, take-off and approach transitions, high-speed level flight, autorotation, and stabilator bias actuator hardovers. In the simulation study, stabilator design was varied to establish trends due to design changes and to evaluate potential stabilator design modifications for improved handling qualities. Design changes included stabilator area, aspect ratio, bias angle, and coupling ratio.

For the conditions tested during hovering flight in and out of ground effect, the stabilator in the "fly" (horizontal) mode did not produce the pitching oscillations that were experienced by the S-61F (NH-3A) research helicopter. The stabilator is located so that no adverse disturbances were experienced in steady hover in either the "fly" or "free" mode. In fact, the "free" mode was not required for the conditions tested. Also, no unusual characteristics were observed in paced sideward, rearward and forward low-speed flight. The "free" mode of the stabilator does not offer enough improvement in handling qualities, for the loading condition tested, to justify the additional control system complexity.

The existing stabilator design will provide the aircraft with acceptable trim and static stability characteristics for the full gross weight and center of gravity envelope. The in-flight trimmability of the stabilator provides control of fuselage attitude independent of the rotor. However, one trim bias angle of +2.5 degrees (leading edge up) is the optimum angle for all loading conditions tested with the aircraft in a clean configuration. Bias angle has no significant effect on the autorotation characteristics. On the S-67, in-flight trimmability of the stabilator provides only limited benefits in either high or low speed flight.

Adequate delay times and sufficient control power are available for recovery from bias actuator hardovers. Rotor control loads, however, will exceed endurance limits with hardovers above 160 knots.

FOREWORD

This report presents results of a research flight test and computer simulation investigation of the effects of a stabilator on the flight and handling characteristics of the S-67 aircraft. This program is part of a four-phase investigation of the flight characteristics of the S-67 aircraft as a representative high-speed winged helicopter design. Evaluation of speed brakes, aircraft maneuverability and a Feel Augmentation System (FAS) are also part of the flight investigation of the S-67 aircraft. FAS provides "force feel" in pitch.

The work was performed by the Sikorsky Aircraft Division of United Aircraft for the U. S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, under Contract DAAJ02-71-C-0010 (DA Task 1F163204D15704). Mr. R. C. Dumond was the Army's Technical Representative.

v

CONTENTS

LIST OF ILLUSTRATIONS

LIST OF SYMBOLS

A_{1_s} main rotor lateral cyclic pitch blade angle measured in shaft axis system, deg

a_t three-dimensional lift curve slope of horizontal tail

a_o two-dimensional lift curve slope

B_{1_s} main rotor longitudinal cyclic pitch blade angle measured in shaft axis system, deg

b_t horizontal tail span, ft

C_s stabilator control coupling ratio, rate of change of stabilator angle with longitudinal cyclic pitch blade angle, (di_t/ dB_{1_s})

C.G.   center of gravity

$d\varepsilon/d\alpha_f$ rate of change of wing and main rotor downwash angle at horizontal tail with fuselage angle of attack

i_t stabilator incidence angle, angle between fuselage reference line and stabilator chordline, deg

i_{t_b} stabilator bias angle, deg

i_{t_o} initial stabilator incidence angle, deg

LED   leading edge down

LEU   leading edge up

l_t tail moment arm, horizontal distance from aircraft C.G. to stabilator aerodynamic center, ft

M_{α_f} rate of change of total pitching moment about aircraft C.G. with fuselage angle of attack, ft-lb/rad

M_{α_t} rate of change of pitching moment about aircraft C.G. due to horizontal tail with fuselage angle of attack, ft-lb/rad

N yawing moment about aircraft C.G., ft-lb

q free-stream dynamic pressure, lb/sq ft

q_t local dynamic pressure at horizontal tail, lb/sq ft

q_t/q horizontal tail efficiency

S_t stabilator area, ft^2

V	forward speed, kn
V_c	rate of climb, ft/min
V_s	sideward airspeed, kn
β	fuselage sideslip angle, deg
β_V	vibratory component of main rotor blade flapping, deg
Θ_{cuff}	main rotor collective pitch blade angle measured at the blade cuff, deg
Θ_f	fuselage pitch attitude, deg
Θ_{TR}	tail rotor impressed collective pitch blade angle, deg
Θ_0	main rotor collective pitch blade angle measured at the center of rotation, deg
$\dot{\Theta}_f$	fuselage pitch angular velocity, deg/sec
$\Delta\Theta_f$	change in fuselage pitch attitude during a maneuver, deg
$\Delta\dot{\Theta}_f$	change in fuselage pitch angular velocity during a maneuver, deg/sec
ϕ_f	fuselage roll attitude, deg

INTRODUCTION

An all-movable, in-flight-trimmable, horizontal stabilizer, called a stabilator, could offer improved performance, handling qualities, and maneuverability. In preliminary design studies, this horizontal stabilizer configuration has shown sufficient improvement in these areas to justify its installation on the S-67 winged helicopter. Some of the attributes of the stabilator are:

- Reduced vertical drag in hover

- Improvement in handling qualities throughout the speed range

- Reduced rotor control loads

- Reduced main rotor flapping and aircraft vibrations

To realize the flying and handling qualities advantages, the S-67 stabilator was designed with the following features:

- Coupled to the longitudinal cyclic stick

- Uncoupled and "free-floating"

- Trimmable in flight

The flying and handling quality advantages of each of these features have been demonstrated in previous research flight tests and analytical studies. For example, an elevator coupled with the longitudinal cyclic stick on the S-61F (NH-3A) research compound helicopter improved the aircraft's handling qualities and reduced main rotor loads, thereby allowing increased maneuverability.[1] A free-floating horizontal tail significantly reduced vertical drag in hover for the HR2S-1 (S-56) Marine Assault Helicopter. In-flight trimming of the elevator on the S-61F (NH-3A) helicopter by a beeper control reduced main rotor flapping and aircraft vibrations by controlling fuselage pitch attitude independent of the rotor.[2]

Although these research programs have evaluated some features of a stabilator, additional research and development is required to evaluate all the features integrated into one design. Some of the questions that need to be answered are:

1. What is the effect of free-floating the stabilator on the handling qualities of the helicopter in hover and low-speed flight?

2. When is the best time to couple the stabilator to the longitudinal cyclic stick in transition from hover to forward flight and to uncouple it during approaches to hover?

3. What is the effect of control coupling and in-flight trimmability on static trim performance and structural loads?

1

4. Will control coupling and in-flight trimmability enhance or degrade autorotation capability?

5. Can the pilot maintain control of the aircraft in the event of a hardover in the bias angle control system?

A research program, sponsored by the U. S. Army Air Mobility Research and Development Laboratory, Eustis Directorate, was carried out to answer these questions. In particular, the objective was to determine if the stabilator offers sufficient advantages to warrant consideration for missions requiring appreciable fuselage attitude control as well as increased speed and maneuverability.

This research program was conducted in two phases: (1) a flight test evaluation of the S-67 stabilator and (2) a computer simulation evaluation of design variations. The flight tests were conducted for several combinations of gross weight and center of gravity locations under the following flight conditions:

. Hover and hovering turns

. Low-speed fore and aft and sideward flight

. Takeoff and approach transitions

. Trimmed high-speed flight

. Autorotation, entry and steady

. Bias angle actuator hardovers in high-speed flight

The effects of changes in design on high-speed static trim and longitudinal static stability were evaluated in the computer simulation phase. The design changes included stabilator area, aspect ratio, bias angle, and coupling ratio as well as gross weight and center of gravity location. The results from flight test and the computer simulation are analyzed to assess the advantages and to establish the operational techniques of the S-67 stabilator.

2

SCOPE OF PROGRAM

A research flight test and computer simulation program investigated the effects of a stabilator on the flight and handling characteristics of the Sikorsky S-67 helicopter. The flight test phase investigated the effects of: (1) coupling and uncoupling the stabilator from the longitudinal cyclic stick in hover, low-speed flight, and takeoff and approach transitions; (2) stabilator bias angle in high-speed flight and autorotation; and (3) stabilator bias actuator hardovers in high-speed flight. The computer simulation phase investigated stabilator design changes on trim and static stability characteristics in high-speed flight.

Listed below are the combinations of gross weight and center of gravity locations that were flown. A forward center of gravity, light gross weight was not flown because it could only be obtained at a very low fuel load. However, this loading condition was investigated in the simulation study.

Loading Condition	Gross Weight (lb)	C.G. Location (in.)
1	17,300	258
2	14,800	276
3	17,300	276

The stabilator bias angle, airspeed, and stabilator coupling mode for each loading condition investigated in the flight test program are listed in Table I. In the uncoupled mode, the stabilator free-floated.

During the hover, hovering turns, and low-speed flight tests, the aircraft was flown at an altitude of 30 feet. The hovering turns were ±5-second turns, both left and right, with the center of rotation at the pilot's seat. Each turn was initiated with the aircraft heading into the wind. The low-speed fore and aft and sideward flight tests were flown at calibrated airspeeds determined by a pace truck and local wind conditions.

The transition flights were takeoff and approach maneuvers. The takeoffs were initiated from hover with the stabilator in the "free" mode. The aircraft was then accelerated to 80 knots, and the stabilator was coupled at the airspeeds listed in Table I. During the approaches, the aircraft was decelerated from 80 knots to a flare and hover. The stabilator was uncoupled from the "fly" mode at the same airspeeds that it was coupled in the takeoffs.

Data were recorded in high-speed flight from 80 knots to maximum level-flight speed. The full range of stabilator bias angles from +5 degrees (leading edge up) to -5 degrees (leading edge down) was evaluated at the forward C.G. condition (Condition 1).

The autorotations were initiated from 80- and 100-knot trim speeds with the stabilator coupled. Both entries and stead￼y autorotative descents were evaluated at the bias angles indicated in Table I.

Stabilator bias actuator hardovers, leading edge up and down, were

3

Flight Condition	Airspeed (kn)	Stabilator Coupling	Stabilator Bias Angle (deg)		
			Loading Condition 1	Loading Condition 2	Loading Condition 3
Hover	0	C/U*	-2.5	-	2.5
Hovering Turns	0	C/U	-2.5	-	2.5
Forward Flight	10,20,30	C/U	-2.5	-	2.5
Rearward Flight	-5, -10, -15, -20	C/U	-2.5	-	2.5
Sideward Flight	±5, ±10, ±15, ± 20	C/U	-2.5	-	2.5
Takeoff and Approach	20,30,40 50,60	C/U	-2.5	2.5	2.5
High-Speed Flight	80,100,140, 160,170,V_{max}	C	0,+2.5, +5.0	0,2.5, 5.0	0,2.5,5.0
Autorotation	80, 100	C	0,-2.5, -5.0	-	0,2.5,5.0
Bias Actuator Hardover	60, 100, 140,160	C	-	-	2.5,+2.5, 2.5,-7.5

TABLE I.   FLIGHT TEST CONDITIONS

* Coupled/Uncoupled

evaluated at flight speeds from 60 to 160 knots for loading Condition 3. The bias control actuator hardover tests were conducted by increasing/ decreasing the bias angle from the initial trim value to the upper/lower limits of the bias angle range at a rate of 0.14 deg/sec. Thus, from the trim setting of 2.5 degrees, bias angle was alternately increased 2.5 degrees to the leading-edge-up limit and decreased 7.5 degrees to the leading-edge-down limit.

Flight test results were used to verify and update the S-67 Computer Simulation Program for the simulation study. In this study, gross weight, center of gravity location, stabilator area, aspect ratio, bias angle, and control coupling ratio were varied over the ranges shown in Table II. Trimmed flight with zero roll angle were simulated at speeds from 60 knots to maximum level-flight speed to establish design change trend lines on static trim and longitudinal static stability. In the static stability study, the aircraft was initially trimmed at each of the speeds shown in Table II. Forward speed was then changed in increments of ±5 and ±15 knots by adjusting longitudinal control. Collective control and rotor RPM were held constant. Lateral and directional controls were re-trimmed at each airspeed increment to reduce the cross-coupling effects.

5

Gross Weight (lb)	C.G. Location (in)	Stabilator Area (ft²)	Stabilator Aspect Ratio	Stabilator Control Coupling Ratio	Stabilator Bias Setting (deg)	Forward Speed (kn)
TABLE II. COMPUTER SIMULATION STUDY CONDITIONS						

STATIC TRIM

Gross Weight (lb)	C.G. Location (in)	Stabilator Area (ft²)	Stabilator Aspect Ratio	Stabilator Control Coupling Ratio	Stabilator Bias Setting (deg)	Forward Speed (kn)
17,300 ↓ 22,000 ↓	258/276 ↓	50 25 75 50 ↓	4.8 ↓ 2.0 6.0 4.8 ↓	1.0 ↓ 0 0.5 1.5 1.0 ↓	2.5 ↓ 5.0 0 -5.0	60,80,100, 140,160,V_{max} ↓

STATIC STABILITY

Gross Weight (lb)	C.G. Location (in)	Stabilator Area (ft²)	Stabilator Aspect Ratio	Stabilator Control Coupling Ratio	Stabilator Bias Setting (deg)	Forward Speed (kn)
14,800 17,300 22,000 17,300 ↓	258/267/276 ↓ 276 ↓	50 ↓ 25 75 50 ↓	4.8 ↓	1.0 ↓ 0 0.5 1.5 1.0 ↓	2.5 ↓ 5.0 0 -2.5 -5.0	100,140,160* ↓

* Speed was varied ±5 and ±15 knots about each trim speed.

6

DESCRIPTION OF AIRCRAFT

The S-67 demonstrator aircraft is a high-speed derivative of the Sikorsky S-61 (SH-3D) helicopter. The aircraft is shown in three-quarter left view in Figure 1. The narrow, low-drag airframe was designed to meet the high-speed requirements of the attack mission. The cockpit is arranged in tandem with the copilot-gunner in the forward seat and the pilot in the aft, elevated seat. The pilot has visibility down to minus 15 degrees over the nose. Two T58-GE-5 engines are mounted in the main rotor pylon above the fuselage center section.

The main rotor hub, tail rotor, drive system, and transmission systems are all SH-3D dynamic components. The main rotor has five S-61F blades each with a twist of -4 degrees. The 22-inch blade tips are swept back 20 degrees to delay tip Mach number effects. The rotor control system uses SH-3D components and the CH-54 automatic flight control system.

The fixed-wing type control surfaces include the stabilator, a vertical stabilizer, and sponsons with stub wings. The vertical stabilizer is fixed. The tail wheel is attached to the base of the lower, ventral fin, and the retractable main landing gear is housed in the wing sponsons. Wings are attached to the sponsons for additional lift and attachment points for armament. The wing panels have speed brakes to control dive angle and increase deceleration capability.

Principal dimensions and general data for the S-67 aircraft are as follows:

Main Rotor

Diameter	62 ft
Normal Tip Speed (104 percent N_R)	686 ft/sec
Disc Area	3019 ft^2
Solidity	0.0781
Number of Blades	5
Blade Chord	1.52 ft
Blade Twist	-4 deg
Airfoil Section	NACA 0012 MOD
Articulation	Full Flapping and Lagging
Tip Sweep	20 deg

Tail Rotor

Diameter	10 ft 4 in.*
Tip Speed	700 ft/sec
Disc Area	83.9 ft^2

---

*Diameter has been increased 3 in., to 10 ft 7 in.

Tail Rotor (cont'd)

Solidity	0.1885
Number of Blades	5
Blade Chord	0.612 ft
Blade Twist	0 deg
Airfoil Section	NACA 0012 MOD
Pitch Flap Coupling	45 deg

Fuselage

Overall Length	64 ft 1 in.
Overall Height	16 ft 3 in.
Overall Width	27 ft 4 in.
Wheel Tread	7 ft
Wheel Base	36 ft 2 in.

Stabilator

Root Chord	4 ft 2 in.
Tip Chord	2 ft
Taper Ratio	0.48
Area	50 ft^2
Span	15 ft 6 in.
Aspect Ratio	4.8
Airfoil (Root)	NACA 0015
Airfoil (Tip)	NACA 0012

Vertical Fin

Root Chord	7 ft 6 in.
Tip Chord (Upper)	2 ft 10 in.
Tip Chord (Lower)	3 ft 9 in.
Taper Ratio (Upper)	0.62
Taper Ratio (Lower)	0.5
Total Area	68.7 ft^2
Aspect Ratio	2.65
Airfoil Section	NACA 4415

Wing

Root Chord	4 ft 6 in.
Tip Chord	1 ft 11.5 in.
Overall Span	27 ft 4 in.

<u>Wing</u> (cont'd)

Total Exposed Area	58 ft^2
Incidence	8 deg
Dihedral	10 deg
Quarter Chord Sweep	10 deg 45 min
Taper Ratio (Exposed)	0.44
Aspect Ratio	8.0
Airfoil Section, Root	NACA 4415
Airfoil Section, Tip	NACA 4412

<u>Propulsion System</u>

Engines	Two T58-GE-5
Takeoff Power (each)	1500 HP
Military Power	1400 HP
Normal Power	1250 HP
Transmission Rating	2800 HP

<u>Loading Conditions</u>

*Empty Weight	10,900 lb
Maximum Gross Weight Flown	18,000 lb
Maximum Gross Weight Capability	21,800 lb
Center of Gravity Range	258 in. to 276 in.

---

* Aircraft less fuel, payload and crew

HELICOPTER AND STABILATOR FLIGHT CONTROLS

DEFINITION OF TERMS

Coupled Controls: In the coupled, or "fly", mode the stabilator is connected to the pilot's and copilot's longitudinal cyclic stick. The longitudinal stick, then, controls both the stabilator angle and the longitudinal cyclic pitch blade angle, B_{1_s}, of the main rotor. The lateral and collective sticks control the lateral cyclic pitch, A_{1_s}, and collective pitch, θ_o, blade angles respectively. The pedals control tail rotor collective pitch blade angle, θ_{TR}. Views of the stabilator in the "fly" mode are shown in Figures 1 and 2.

Uncoupled Controls: In this mode the stabilator is disconnected from the longitudinal cyclic stick and free-floats. The pilot's and copilot's cockpit controls affect only the main and tail rotor blade angles. Views of the stabilator in the uncoupled, or "free", mode are shown in Figures 3 and 4.

Incidence Angle: The angle between the stabilator chord line and a reference fuselage water line is defined as the incidence angle. The stabilator angle is the total incidence angle of the stabilator. Stabilator leading edge up constitutes a positive, +, incidence angle.

Bias Angle: The incidence angle trimmable by the pilot is the bias angle. Bias angle is controllable only in the coupled mode.

Coupling Ratio: The ratio comprised of degrees of stabilator incidence angle per degree of longitudinal cyclic pitch blade angle is the stabilator control coupling ratio. The relationship between stabilator angle, i_t, initial incidence, i_{t_o}, bias, i_{t_b}, coupling ratio, C_s, longitudinal cyclic pitch blade angle in degrees is given by the following equation:

$$i_t = i_{t_o} + i_{t_b} + C_s B_{1_s}$$

HELICOPTER CONTROL SYSTEM

The S-67 flight control system is similar to the SH-3D helicopter control system. The pilot control motions are fed by push rods to the auxiliary servos. Output motions from the servos are transmitted to a mixing unit where collective control is mixed with lateral cyclic and tail rotor collective controls. Longitudinal control is not mixed with collective Push rods transfer the collective, longitudinal cyclic, and modified lateral cyclic inputs to the main rotor blade pitch servos. Cables carry the modified directional controls to the tail rotor.

STABILATOR CONTROL SYSTEM

The stabilator controls are connected to the longitudinal cyclic stick at the output side of the auxiliary servo. The control commands are transmitted by push rods to the bias actuator and by bell cranks to the stabilator

control servo as shown in Figure 5. The bias actuator is controlled by
an electric beeper switch on the pilot's and copilot's cyclic stick.
Coupling and uncoupling the stabilator from the longitudinal cyclic stick
is controlled by a toggle switch on the pilot's left-hand auxiliary control
panel. A warning light is located on the caution light panel. In the
uncoupled mode the light is on and backlights the words "stabilizer free".
A stabilator angle indicator on the pilot's instrument console indicates
the stabilator trim bias angle, in degrees, in the coupled mode. There is
no indication of stabilator incidence angle in the "free" mode.

Actuation of the coupling toggle switch from the "free" to the "fly" posi-
tion causes the single-acting hydraulic droop actuator, shown in Figure 5,
to rotate the stabilator leading edge down. A pin then locks the stabilator
to the control actuator. When the toggle switch is moved to the "free"
position, the locking pin is withdrawn electrically and the stabilator free-
floats to the vertical position. However, an automatic airspeed switch will
prevent the pin from withdrawing until airspeed is less than 30 knots. In
decelerating approaches, then, the stabilator cannot be uncoupled to the
"free" mode above this speed. In take-off and accelerating flight, however,
the stabilator can be coupled to the "fly" mode at any speed. During the
approach transition flight tests, the automatic airspeed switch was tempor-
arily bypassed to permit uncoupling tests up to 60 knots.

FLIGHT TEST RESULTS

The flight test results are discussed in the following sequence: (1) hover and low-speed flight - forward and rearward, (2) hover and sideward flight - left and right, (3) hovering turns - 15-second, 360-degree turns - left and right, (4) transition flight - coupling to "fly" mode in accelerating takeoffs and uncoupling to "free" mode in decelerating approaches, (5) high-speed static trim flight, (6) autorotation - entry and trimmed, and (7) stabilator bias actuator hardover - leading edge up and down.

HOVER AND LOW-SPEED FLIGHT

The first step in evaluating the S-67 stabilator was to determine the effect of uncoupling and coupling the stabilator on the performance and handling qualities of the aircraft in hover and low-speed flight. The aircraft was flown at a gross weight of 17,300 pounds for both forward (258 in.) and aft (276 in.) center of gravity loading conditions. In the coupled mode, stabilator bias angle was -2.5° for the forward c.g. condition and $+2.5^\circ$ for aft c.g. All tests were flown in ground effect at an altitude of 30 to 35 feet. One hovering flight was made at a 100-foot altitude to evaluate the effects of uncoupling the stabilator in and out of ground effect.

For all conditions tested, the aircraft was very stable in hover. It was easy to establish trim and excursions from trim developed very slowly. With the stabilator in the "fly" mode, hovering both in and out of ground effect under varying headwind conditions did not produce the pitching oscillations that were experienced with the S-61F (NH-3A) helicopter[3]. In fact, no adverse disturbances were experienced in hover with the stabilator in either the "fly" or the "free" mode. Furthermore, in accelerations from hover to 20 knots with the tail wheel just off the ground, no pitching disturbances were observed. Activation of the stabilator switch from the "fly" to the "free" mode, and vice versa, in stabilized hovering flight with 5- to 8-knot winds, caused an immediate transient pitch response of 4 to 6 degrees. The aircraft, however, returned to the pitch attitude that existed before the switch was activated. No adverse or unusual characteristics were observed in forward, rearward, and sideward flight with the stabilator in the "fly" or "free" mode.

As indicated in Figures 6 and 7 and supported by pilot opinion, the handling qualities of the aircraft are not significantly different with the stabilator in the "free" or "fly" mode. However, there was a difference in fuselage pitch attitude and longitudinal cyclic stick position for the forward c.g. condition in fore-and-aft flight (Figure 6(a)) and for the aft c.g. condition in sideward flight (Figure 7(a)). These differences are not considered to be adverse or unusual.

In low-speed forward flight with increasing forward speed, the main rotor downwash is gradually swept rearward to impinge on the horizontal tail. The resulting download on the stabilator in the coupled mode causes the aircraft to pitch up. The pilot trims this nose-up moment by moving cyclic stick forward (Figure 6(a)). In the uncoupled mode, with the stabilator

12

near vertical, the download is smaller. Thus, a smaller nose-up pitching moment is applied by the stabilator, the aircraft pitch attitude is lower, and less forward stick is required. Because the forward c.g. condition gives a greater nose-down pitch attitude and thus positions the tail more into the main rotor downwash, the uncoupled tail is more effective in reducing tail download, aircraft pitch attitude change, and control stick movement.

In rearward flight the main rotor downwash is swept forward and away from the horizontal tail. The stabilator is trailing-edge-up in the coupled mode and, thus, generates an up-load and nose-down pitching moment which is trimmed with aft stick. When the stabilator is uncoupled, its pitching moment contribution is less, so pitch attitude is less downward and less aft stick is required for trim.

There is no difference in performance in hover and low-speed flight with the stabilator coupled or uncoupled (Figure 6(b)). For the forward c.g. condition, where the advantages of uncoupling the stabilator on stick movement and pitch attitude were more evident, a small decrease in power required occurs in forward flight. In rearward flight, the reduced lift of the uncoupled tail causes the main rotor thrust to increase and the corresponding main rotor power required to be higher. In hovering flight at an altitude of 100 feet with aft c.g., main rotor power was 1552 HP with the stabilator coupled and 1535 HP with the stabilator uncoupled, an insignificant difference.

The trim characteristics in sideward flight (Figure 7) also show no significant effects of uncoupling the stabilator. As the main rotor downwash is swept laterally in sideward flight, its influence on the tail is stronger for the aft c.g. condition. The uncoupled stabilator is more effective in reducing tail download, pitch attitude, and longitudinal stick displacement at this loading condition than for the forward c.g. as shown in Figure 7(a). The pitch-down and aft stick movement that occurs in right sideward flight is due to aerodynamic interference effects between the main rotor and the tail rotor. The reduction in tail download when the stabilator is uncoupled also produced a consistent reduction in main rotor flapping (Figure 7(b)) and main rotor horsepower (Figure 7(c)) for the aft c.g. loading condition.

The results presented in Figure 7 (lateral cyclic stick position was not available) show that less roll angle and rudder pedal movement are required in sideward flight to maintain a constant heading with the stabilator uncoupled. This reduction in roll angle and pedal movement is more pronounced with a forward c.g. condition.

HOVERING TURNS

The effect of stabilator mode on the hovering turns of the S-67 aircraft was investigated by flying 15-second, 360-degree hovering turns to the left and right. The turns were performed in ground effect at 30-35 feet altitude, with the aircraft rotating about the pilot's seat. The aircraft weighed 17,300 pounds for both forward and aft c.g. positions. The turns were executed with the stabilator coupled and uncoupled and were initiated with the aircraft headed into the wind. 13

Since the stabilator is a longitudinal trim and control device, the total longitudinal stick displacement and pitch attitude change during the turn determine the effects of the stabilator mode on hovering turns. The results with the stabilator coupled and uncoupled for both c.g. conditions are shown in the form of bar graphs in Figure 8. Longitudinal stick travel and pitch attitude change during the turns are reduced by uncoupling the tail for the forward c.g. condition. For aft c.g., there was very little difference due to stabilator mode.

Thus, the performance and handling qualities of the aircraft in hover and hovering turns do not vary significantly with stabilator mode. Pitching oscillations in hover, which were experienced by the S-61F (NH-3A) research helicopter, did not occur in either mode. Also, no adverse or unusual characteristics were observed in sideward, rearward, and forward low-speed flight.

TRANSITION FLIGHTS

Takeoff and approach transition flights were conducted to develop piloting techniques in transition and to select the best speed for coupling and uncoupling the stabilator. In the takeoff flights, the aircraft was hovered and then accelerated to 80 knots. Successive flights were made with the stabilator coupled at hover, 20, 30, 40, 50, and 60 knots. The procedure was reversed in the approaches to hover.

The S-67 stabilator, when uncoupled and as originally installed on the aircraft, demonstrated, in ground and flight tests, freedom to rotate from -25° (leading edge down) to $+100^\circ$ (leading edge up). Prior to the stabilator flight evaluation, the stabilator and control system were modified to assure freedom from torsional divergence in high-speed flight. Following these modifications, the stabilator in the free mode tended to remain at arbitrary incidence angles in low-speed flight due to friction forces in the stabilator control system. The aerodynamic hinge moments were too small, at times, to rotate the stabilator. Because of this stabilator "hang-up", different aircraft responses occurred following stabilator coupling and uncoupling for the same conditions. A time history plot of aircraft and control response in transition flight is presented in Figure 9. The stabilator was coupled at 40 knots during an accelerating takeoff from hover. Gross weight is 17,300 pounds; center of gravity is forward at 258 inches; and stabilator bias angle in the "fly" mode is -2.5 degrees. This test condition illustrates the large control and aircraft response that accompanies coupling of the stabilator in forward flight. Although the figure shows that the locking pin was engaged at 9.7 seconds, it was not known when the pilot moved the switch to the "fly" mode. However, from pitch rate and attitude information, it is estimated that the locking pin was engaged 2 seconds after the selector switch was activated - a 2-second delay time. For an OGE hover, this delay time can be as long as 4 seconds.

The nose-up moment generated by coupling the stabilator at 40 knots produced a change i pitch angular rate of 18°/sec and rotated the aircraft

14

nose upward about 14.7°. The pilot responded with a forward stick movement of 51.6% of allowable travel - a control displacement of 7.2 inches. The aircraft was stabilized within 2 seconds after the stabilator was locked or 4 seconds after the switch was moved to the "fly" mode.

The primary response in changing the stabilator mode is in pitch. Cross-coupling effects caused some response in roll rate and roll angle, which required only small lateral stick movement to control (Figure 9). There-fore, only the longitudinal control and response time were used to evaluate the effect of coupling and uncoupling the stabilator during transitions.

Although large stick motions were required during the takeoff accelerations to counteract the pitching disturbances, adequate control margins were available (Figure 10). The pilots tended to overshoot to control the angular acceleration and rate. The most forward stick position reached was 85%. Although the pitch-up tendency was abrupt, pitch attitude never exceeded a nose-up value of 15 degrees. The stick activity and pitching motion were much less in the approaches (Figure 11). The most aft stick position was 15%, and maximum stick movement was about 28% or 4 inches.

The difference between the takeoffs and approaches is more apparent in Figure 12. The stick and aircraft motions in takeoffs are approximately twice as large as in approaches. For both flight conditions, the pitch response and stick motion are nearly proportional to the square of the airspeed. A comparison of the curves for gross weight and center of gravity position indicates that the disturbance and control motion will be larger for the lighter gross weight and forward c.g. conditions. The moments of inertia are smaller at the lighter weight, and the tail moment arm is larger with c.g. forward. Both of these factors will produce larger dis-turbances and will require larger stick movement for recovery.

Maximum level-flight speed with the stabilator uncoupled should not exceed 20 knots. During takeoffs, with the stabilator uncoupled, forward speed should be increased gradually. As the stabilator becomes effective, a distinct nose-down tendency occurs. During these tests, 2 to 3 inches of rearward stick movement was required at a 40-knot coupling speed to keep the aircraft nose up and to maintain acceleration. This stick reversal was objectionable. Therefore, the stabilator should not be coupled above 20 knots. At forward speeds, even as low as 20 knots, the free-floating tail introduces spurious disturbances which are undesirable. Furthermore, pilot activity and concentration is high during the pushover into forward acceleration and during the flare to hover. It is recommended, therefore, that in takeoff transition flight, the stabilator should be coupled in hover before the aircraft is rotated for acceleration to forward flight. In approaches, the stabilator should be kept in the "fly" mode until the aircraft is in a hover. In fact, for all hover and low-speed flight condi-tions tested, the "free" mode was not a requirement.

FORWARD FLIGHT

Level forward flight tests were conducted to select the optimum bias angle trim setting for each loading condition, and to check the stabilator for

flutter characteristics. Airspeeds from 80 knots to maximum level-flight speed were investigated for the following loading combinations: fore and aft c.g. at 17,300 pounds and aft c.g. at 14,800 pounds, load conditions 1, 2, and 3. As explained in the Scope of Program section, the forward c.g., 14,800-pound loading condition was impractical to fly.

A bias angle of +2.5 degrees was recommended, from previous simulation studies, as the value for the aft c.g. conditions. A value of -2.5 degrees was recommended for forward c.g. These values of bias angle were varied in 2.5-degree increments to determine the effect of changes in bias angle and to select the optimum angle for trimmed flight. However, results for the 17,300 pounds, forward c.g. condition indicated that with a bias angle of -5 degrees, cyclic stick reached its forward stop well below the maximum speed capability of the aircraft. Therefore, the full bias angle range ($\pm 5^\circ$, $\pm 2.5^\circ$, and 0°) was investigated at this loading condition.

On fixed-wing aircraft, horizontal tail trimming devices are used primarily to reduce stick forces. On helicopters, the trimmable stabilator is a means of controlling both longitudinal cyclic stick position and fuselage pitch attitude independent of the rotor (Figures 13(a) and 14). In cruise flight, going from a -5-degree bias angle to a +5-degree bias angle can cause a 20% reduction in forward stick position. This is equivalent to a 3-inch reduction in stick displacement, providing the pilot with a more comfortable position and increased stick margins for maneuvers. The advantages of reduced stick motion are achieved, however, at the expense of larger nose-down attitudes. The design tradeoff, then, is to select the optimum bias angle which will provide (1) adequate stick margins for maneuvers and (2) desirable nose-down pitch attitudes for minimum fuselage drag and wing/fuselage download.

Since stabilator bias angle is trimmed in the longitudinal mode, little or no effects in the lateral and yaw modes occurred. Lateral cyclic stick position (Figure 13(b)) is independent of bias angle. Fuselage roll attitude (Figure 13(b)) and sideslip angle (Figure 13(c)) show small but uniform effects of bias setting. With positive bias angle, more of the vertical tail area leaves the flow of the main rotor downwash and enters free-stream dynamic pressure. The cambered vertical tail unloads the tail rotor and reduces the required tail rotor blade angle (Figure 13(c)). More right pedal travel is required. As with the lateral cyclic stick position, main rotor shaft horsepower and collective stick position (Figure 13(d)) are independent of bias angle.

Stabilator angle and main rotor flapping (Figure 13(e)) which vary directly with longitudinal cyclic stick position, show large changes with bias angle. The forward stick positions with negative bias angle (Figure 13(a)) cause a larger tip path plane tilt (and flapping) than the positive settings. For this loading condition, a bias angle of +2.5° produces a lower value of flapping. With negative bias angle, longitudinal stick position is forward so that the tip path plane tilts down over the nose of the aircraft. As the stick moves back with positive bias angle, the tip path plane rotates rearward and down over the tail. For this reason, the curves for positive bias angle have a different shape than for negative angles, and flapping is

a minimum for a bias value of +2.5° (Figure 15).

Main rotor shaft vibratory bending stress (Figure 13(f)) varies with speed. Minimum stress occurs at +2.5° bias angle and is 2000 psi, which is well below the endurance limit of ±18,600 psi. The advantage of using a stabilator to reduce shaft bending stress is illustrated by Figure 13(f). At 150 knots, shaft bending stress can be varied by 7000 psi over the full bias angle range.

In forward flight, stabilator bias angle can control pitch attitude and wing angle of attack. Thus, by adjusting bias angle, the pilot can transfer lift from the rotor to the wing. Rotor stall can thereby be delayed, and rotor control loads can be reduced. The flight test results show, however, that wing lift in cruise flight is not large enough to have a major effect on rotor stall or rotor control loads. As shown in Figures 13(f) and 13(g), main rotor right lateral stationary star load, main rotor push rod load, and stabilator torque tube stress were not significantly affected by stabilator bias angle. Main rotor right lateral stationary star load is the control load measured on the lower, non-rotating main rotor swashplate at the right lateral servo.

Since structural loads are not significantly affected by bias angle, optimum bias angle can be selected on the basis of the following criteria:

1. Stick position at maximum level-flight speed should be 90% or less to provide adequate control margins for high-speed maneuvering.

2. Fuselage pitch attitude should be about 5° nose down. At this pitch angle, the wing provides enough lift to balance the fuselage download, but an angle of attack margin is available for the wing to develop load factor in maneuvers without stalling.

3. Main rotor flapping should be a minimum at maximum forward speed to minimize shaft bending stress and aircraft vibrations, and to assure that flapping limits do not restrict the performance or maneuverability of the aircraft at any point in its flight envelope.

The results in Figures 14 and 15 indicate that for all loading conditions tested, a bias angle of +2.5° is optimum. Fuselage attitude is about 5 degrees nose down, longitudinal cyclic stick position is 90% or less, and flapping, on the order of 2 to 3 degrees, is at or near the minimum. Moreover, at a bias angle of +2.5 degrees there is only a slight reduction in maximum speed capability (Figure 16).

STABILATOR FLUTTER CHARACTERISTICS

During initial flight tests, high stabilator oscillatory stress levels indicated a torsional divergence problem. It was a low, 4 cps, stabilator control system torsional natural frequency. The control system stiffness was increased by installing a hydraulic servo at the stabilator control horn (Figure 5). Also, the stabilator mass moment of inertia was reduced

17

by removal of the tip weights. As a result of these modifications, the system natural frequency was increased to 35-40 cps, which eliminated the problem of torsional divergence.

A flutter analysis of the stabilator system showed that flutter would not occur until 315 knots airspeed. This is well in excess of the 230-knot flutter design speed requirement. Stabilator stresses were monitored in subsequent flight tests, which included maneuvers and speeds over 200 knots. No indication of flutter was observed. The stabilator system is free of flutter within the flight envelope of the aircraft.

AUTOROTATION

Autorotation flights were conducted to evaluate the effects of stabilator bias angle on the autorotation characteristics of the S-67 aircraft. The evaluation included autorotation entry maneuvers and steady-state auto-rotations for 80- and 100-knot entry airspeeds with forward and aft C.G. loadings at 17,300 pounds. Forward speed in steady autorotation is limited to 105 knots for the S-67 aircraft by the minimum allowable rotor rpm of $96\% N_R$.

Although it was effective in high-speed trimmed level flight, the trimmability feature of the stabilator has no significant effect on the autorotation characteristics of the aircraft. In the autorotation entry maneuver stabilator bias angle can reduce the nose-down pitching motion caused by the rapid lowering of collective stick (Figure 17). However, in steady autorotation rotor RPM is independent of bias angle (Figure 18). Also, although rate of descent showed a measurable difference with bias angle, the differences are insignificant. At the low forward speeds in autorotation, 80 to 100 knots, wing lift was small, so rotor lift load and rate of descent were not significantly reduced by variations in bias angle. Also, no significant reduction in stick displacement from trimmed level flight to steady auto-rotation was achieved by varying stabilator bias angle. Fifteen percent (2 inches) of aft stick movement is required to establish autorotation (Figure 13(a) and 18).

BIAS ACTUATOR HARDOVERS

Stabilator bias angle actuator hardover tests were conducted to determine:

- . Aircraft attitudes and motion resulting from the hardover

- . Delay times available before recovery action is required

- . Available control power for recovery

- . Loads on the aircraft during the disturbance and recovery

The tests were conducted at 60, 100, 140, and 160 knots with the aircraft at the heavyweight (17,300 pounds) aft c.g. condition. Trim bias angle was +2.5 degrees. Bias hardovers in both directions were investigated

from this trim angle. Since the bias angle limits are ±5 degrees, the changes were +2.5 degrees leading edge up and -7.5 degrees leading edge down. The bias actuator rotates the stabilator through its 10-degree bias range in 7 seconds. The aircraft was first trimmed in level flight, and then the bias control switch on the cyclic stick was depressed and held until the upper/lower limit was reached. Corrective control action was delayed until pitch attitude approached 30 degrees.

The 2.5-degree change in bias angle leading edge up was mild (Figures 19 and 20). At 60 knots, recovery action was delayed more than 10 seconds. At 160 knots, the delay time was limited to 1 second to avoid exceeding main rotor control load endurance limits. Scheduled tests at maximum forward speed, 180 knots, were deleted because of the high control loads. However, at all test airspeeds, adequate control power was available to recover from the leading-edge-up hardover and aircraft attitudes were not excessive.

A hardover in the opposite direction, 7.5-degree leading edge down, was, by comparison, a severe disturbance (Figure 21). However, adequate control power was still available. Maximum longitudinal cyclic stick position was below 90%, and adequate delay time was available (1 second or more) for recovery. Maximum pitch and roll attitudes reached 35 degrees and 60 degrees, respectively. Maximum load factor (1.91) was below the design load factor of 3.3.

Adequate delay time was available at all speeds for recovery from the bias hardover. At forward speeds of 140 knots and higher, the leading-edge-down bias hardover caused the main rotor control loads to exceed endurance within the first 1 to 1.5 seconds. At 160 knots, the aircraft motion became divergent which is typical of the aft c.g. loading condition. Adequate control power was available at all speeds, however, to recover without reaching limiting or dangerous aircraft attitudes. Also, the initial aircraft response to the hardover was not severe. From 1 to 1.5 seconds was required before pitch angular acceleration became noticeable to the pilots. A bias control actuator hardover will not unduly jeopardize safety-of-flight. During a stabilator leading-edge-down hardover above 160 knots, rotor control load endurance limits will be exceeded.

COMPUTER SIMULATION RESULTS

A computer simulation study was conducted, in conjunction with the flight tests, to show the effect of stabilator design changes on aircraft stability and control characteristics. A detailed description of the computer program and correlation with flight test results are presented in Appendix II.

The primary requirement of the S-67 stabilator is to provide the aircraft with an acceptable level of inherent stability. The design criteria are:

> stable longitudinal cyclic stick trim position gradients for all loading conditions, and

> neutral static stability at 165 knots cruise speed for aft c.g. loading conditions.

Static stability requires, at least, that in response to a disturbance in angle of attack, pitching moments are generated which tend to return the aircraft to its equilibrium condition. The aircraft, therefore, must have negative angle of attack stability, $M_{\alpha_f} < 0$. The contribution of the stabilator to angle of attack stability, per unit dynamic pressure, is given by the following expression:

$$M_{\alpha_t}/q = -a_t \, S_t \, l_t \, (1 - d\varepsilon/d\alpha_f) \, (q_t/q)$$

Thus, the key parameters which influence the stability contribution of the stabilator are:

Lift Curve Slope, a_t

Area, S_t

Moment Arm, l_t

Tail Efficiency, q_t/q

Main Rotor and Wing Downwash Effects, $d\varepsilon/d\alpha_f$

The stabilator lift curve slope is a function of its aspect ratio and section lift curve slope, a_o. The section lift curve slope in turn depends on the airfoil section. For 12 to 15% thick airfoils, selected to meet aerodynamic and structural requirements, the differences in a_o may be neglected. Thus, aspect ratio controls the lift curve slope.

To investigate the effects of stabilator design changes, stabilator area, aspect ratio, bias angle, and coupling ratio were varied as shown in Table II. Variations of tail moment arm, l_t, were evaluated by varying c.g. position. However, the effects of the tail location relative to the wing, which establish the wing downwash effects as well as the tail efficiency, q_t/q, were incorporated experimentally through the wind tunnel data. Planform variations, such as taper ratio and twist, which affect the spanwise

20

load distribution and tip stall, are beyond the scope of this study.

STATIC TRIM

The effects of stabilator design changes on the longitudinal trim charac-
teristics of the aircraft are presented in Figures 22 through 26. In each
figure the effects on longitudinal cyclic stick position, fuselage pitch
attitude, and wing/fuselage lift are presented as a function of forward
speed.

Stabilator Area

Although stabilator area is determined by static stability considerations,
the design area of 50 ft^2 provides acceptable longitudinal trim (Figure 22).
Stick trim position gradients are stable for all loading conditions and
adequate wing angle of attack and control margins are available for high-
speed maneuvering. As shown in Figure 22, longitudinal trim is affected
more by c.g. position than stabilator area. With the c.g. in the forward
position, stabilator area has no significant effect on trim in cruise and
high-speed flight, (Figure 22(a)). With aft c.g., however, a reduction
in stabilator area to 25 ft^2 requires less fuselage nose-down pitch attitude
and more forward stick displacement for trim, which reduce wing angle of
attack and control margins for high-speed maneuvering (Figure 22(b)). These
margins are increased with a 75 ft^2 area, but the slight improvement does
not justify the increase in stabilator weight.

Aspect Ratio

A change in the stabilator aspect ratio from the design value of 4.8 to 6
has little effect on the stabilator lift curve slope and no effect on the
longitudinal trim characteristics (Figure 23). A reduction in aspect ratio
to 2, however, reduces the stabilator lift curve slope. Thus, more forward
stick is required, pitch attitude is higher, and more lift is transferred
to the wing. In the design of horizontal stabilizers, aspect ratios of 3
to 5 are used to obtain a high lift curve slope and, hence, a higher contri-
bution to angle-of-attack stability. The design aspect ratio of 4.8 is a
compromise between aerodynamic and structural considerations. The effects
of aspect ratio are presented for the aft c.g. condition in Figure 23. The
forward c.g. results are similar.

Coupling Ratio

A coupling ratio of 1.0 provides the best trim characteristics (Figure 24).
A coupling ratio of 1.5 will provide additional control power for maneuver-
ing, but is not acceptable because a wing/fuselage download occurs at 180
knots with forward c.g. and a flat stick position gradient occurs for aft
c.g. Coupling ratios less than 1.0 are also not acceptable because adequate
control margins are not available in high-speed flight. A fixed stabilator
incidence, coupling ratio = 0, produces unacceptable trim characteristics.
High nose-up pitch attitudes occur throughout the speed range and excessive
forward stick positions are required.

21

Gross Weight

The existing stabilator design will provide acceptable trim characteristics
for aircraft gross weights up to 22,000 pounds (Figure 25). The increased
rotor thrust at the higher weight produces a higher downwash velocity which
reduces stabilator angle of attack. Although this results in a requirement
for more forward stick, adequate control margins are still available for
trim and maneuvering flight.

Bias Angle

Bias angle was varied at 22,000 pounds gross weight to determine if the
trim characteristics could be further improved. A bias angle of -5 degrees
is not acceptable because of inadequate stick margins for the forward c.g.
condition (Figure 26(a)) and excessive forward stick requirements for the
aft c.g. condition (Figure 26(b)). A bias angle of +5 degrees provides
adequate stick margins, but is not acceptable because the nose-down attitude
reduces the load-sharing contribution of the wing. Although the 0-degree
bias angle provides improved wing load-sharing characteristics, pitch atti-
tude is slightly higher than desirable and less than 10% stick margin is
available at maximum speed for the aft c.g. condition. Thus, a bias angle of
of +2.5 degrees is the optimum trim setting.

STATIC STABILITY

The static stability of the S-67 aircraft was evaluated and the results
are presented in Figures 27 through 33. Airspeed was incremented ±5 and
±15 knots about each of the 100, 140, and 160 knots trim speeds using longi-
tudinal cyclic control. Collective control and rotor speed were held
constant. Lateral stick and rudder pedals were adjusted to trim the lateral
and yaw modes. To establish the incremented airspeeds with constant rotor
RPM, rate of climb was varied.

Stabilator Design

The 50-ft^2 stabilator design provides the aircraft with positive static
stability in high-speed flight (100-160 knots) with the c.g. in the forward
position (Figure 27). With aft c.g., the aircraft has positive static
stability at 100 knots, neutral stability at 140 knots, and negative
stability at 160 knots. The pitch attitude and rate-of-climb gradients are
stable throughout the speed range for all loading conditions.

A strong wing/rotor load-sharing effect with longitudinal cyclic stick
inputs occurred in high-speed forward flight. A rearward stick displacement
produced a rapid increase in rate of climb. This is a result of the negative
static stability of the aircraft in high-speed flight. At the lower speeds,
where the aircraft is statically stable, a 1-inch stick displacement pro-
duces a small change in rate of climb as shown in Figure 27(b). This stick
sensitivity increases rapidly in high-speed flight where the aircraft has
negative stability and the wing is more effective. A rearward stick dis-
placement produces a rapid increase in wing angle of attack and, hence,
wing lift, which increases rate of climb. This control sensitivity is an

22

operational advantage, particularly for maneuvering flight, but increases pilot workload in trimmed flight.

Design Variable Changes

The longitudinal control position gradient for small speed changes about trim is a measure of the static stability of the aircraft. Therefore, it is used to assess changes in the stabilator design. The results of this evaluation are presented in Figures 28 through 31.

With increasing stabilator area (Figure 28), the nose-down moment is larger and more rearward stick displacement is required. Thus, an increase in stabilator area causes the longitudinal control gradient to become more negative and the aircraft to have less static stability. This is also true for changes in bias angle (Figure 29) and control coupling ratio (Figure 30). For desirable stability characteristics, the stabilator must be optimized with respect to combined changes in all three design variables - area, bias angle, and coupling ratio. However, the in-flight trimmability of the stabilator permits the pilot to achieve desirable stability characteristics by adjusting bias angle. For example, the aircraft can be made statically stable at 160 knots if bias angle is reduced from +2.5 degrees to -2.5 degrees, as shown in Figure 29.

The stabilator design, which was optimized for a gross weight of 17,300 pounds, provides the aircraft with acceptable static stability characteristics for a wide range of gross weight variations (Figure 31). The stable characteristics at forward c.g. condition are slightly improved at high gross weights, but the neutral and slightly negative characteristics for the aft c.g. condition are virtually unchanged.

Neutral Point

At the forward c.g. position the aircraft is statically stable at all speeds (Figure 32). As c.g. moves aft, however, the aircraft approaches neutral stability. The fuselage station for this c.g. position is the neutral point of the aircraft. Although military specifications for helicopters[4], do not define neutral point requirements, it is desirable for good handling qualities for the neutral point to be behind the aft c.g. limit.

Since the static stability of the aircraft does not change significantly with gross weight (Figure 31), the change in neutral point is also small (Figure 33). The neutral point is aft of the c.g. envelope up to about 135 knots. Above this speed, the aircraft will display neutral or slightly negative stability characteristics.

The results of this simulation study show that:

1. Stabilator area, bias angle, and coupling ratio have a strong influence on the trim and static stability of the aircraft.

2. The in-flight trimmability of the stabilator provides a means of controlling the trim and static stability characteristics of the

aircraft.

3. No design changes are needed to the stabilator design currently on the S-67 aircraft. It will provide the aircraft with acceptable trim and static stability characteristics for the full gross weight and c.g. envelope.

CONCLUSIONS

The handling qualities in hover of a helicopter configured with a stabilator are dependent on its location relative to the main rotor downwash. The S-67 stabilator location minimizes the main rotor downwash effects and does not cause the pitching oscillations experienced by the S-61F (NH-3A) research helicopter. In hover and low-speed flight for the loading conditions tested, the "free" mode of the stabilator does not offer enough improvement in handling qualities to justify the additional complexity of the control system.

The in-flight trimmability of the stabilator provides control of the fuselage attitude independent of the rotor. However, one stabilator bias angle of 2.5 degrees leading edge up is optimum for all loading conditions tested. Stabilator bias angle has no significant effect on the autorotation characteristics. On the S-67, in-flight trimmability of the stabilator provides only limited benefits in either high- or low-speed flight.

The stabilator design tested will provide the S-67 with acceptable trim and static stability characteristics for its full gross weight and center of gravity envelope.

Adequate delay times and sufficient control power permit recovery from bias angle control actuator hardovers. However, rotor control loads will exceed endurance limits above 160 knots.

LITERATURE CITED

1. Howlett, James J., FLIGHT CHARACTERISTICS OF THE NH-3A COMPOUND HELICOPTER WITH INTEGRATED CONTROL SURFACES, Naval Air Systems Command; Contract N00019-67-C-0513, Department of the Navy, Washington, D. C., November 1969.

2. Reine, C., and Chuga, G., FLIGHT TEST PROGRESS REPORT, S-61F(NH-3A) COMPOUND CONFIGURATION WITH FIVE -4° TWIST BLADES, SER-611145, Sikorsky Aircraft Division of United Aircraft Corporation, Stratford, Connecticut, March 1966.

3. Harper, H.P., and Sardanowsky, W., A STUDY OF TASK PERFORMANCE AND HANDLING QUALITIES EVALUATION TECHNIQUES AT HOVER AND IN LOW-SPEED FLIGHT, Sikorsky Aircraft; USAAVLABS Technical Report 69-47, U. S. Army Aviation Materiel Laboratories, Fort Eustis, Virginia, July 1969, AD 858184.

4. Anonymous, GENERAL REQUIREMENTS FOR HELICOPTER FLYING AND GROUND HANDLING QUALITIES, Military Specification MIL-H-8501A, U. S. Government Printing Office, Washington, D.C., April 1968.

5. Corso, J. J., and Kaplita, T. T., GENERAL HELICOPTER SIMULATION PROGRAM, SER-50542, Sikorsky Aircraft Division of United Aircraft Corporation, Stratford, Connecticut, May 1968.

6. Gifford, J. A., ONE-TWELFTH SCALE WIND TUNNEL TESTS ON THE AH-3 (S-67) DEMONSTRATOR AIRCRAFT, SER-67000, Sikorsky Aircraft Division of United Aircraft Corporation, Stratford, Connecticut, April 1970.

7. Ellison, D. E., and Malthan, L., USAF STABILITY AND CONTROL DATCOM, McDonnel Douglas Corporation; Project No. 8219, Flight Control Division, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, September 1970.
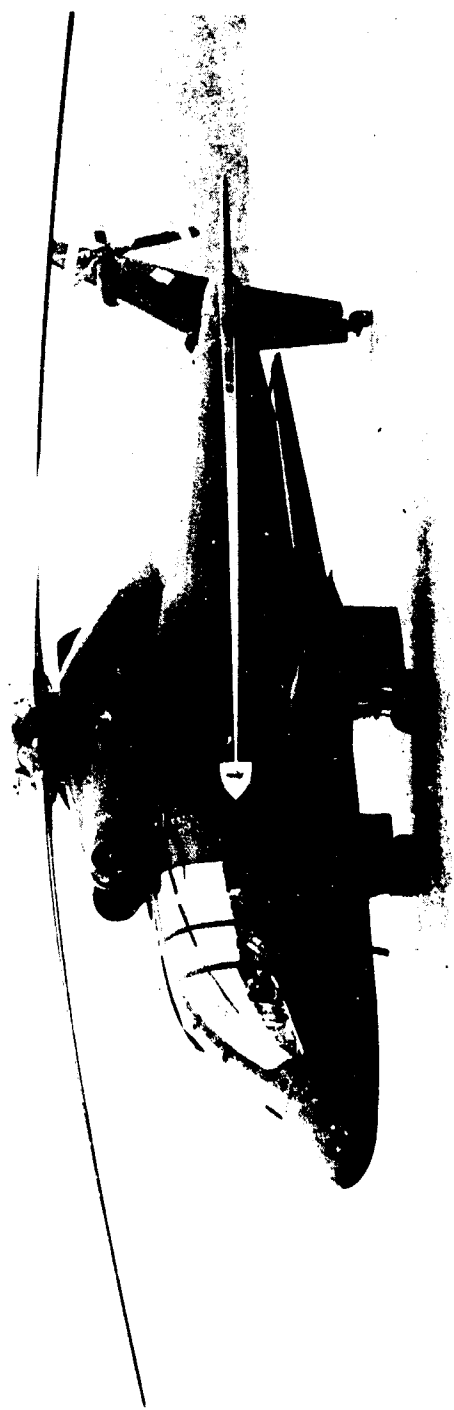
Figure 1. S-67 Aircraft – Three-Quarter Front View.
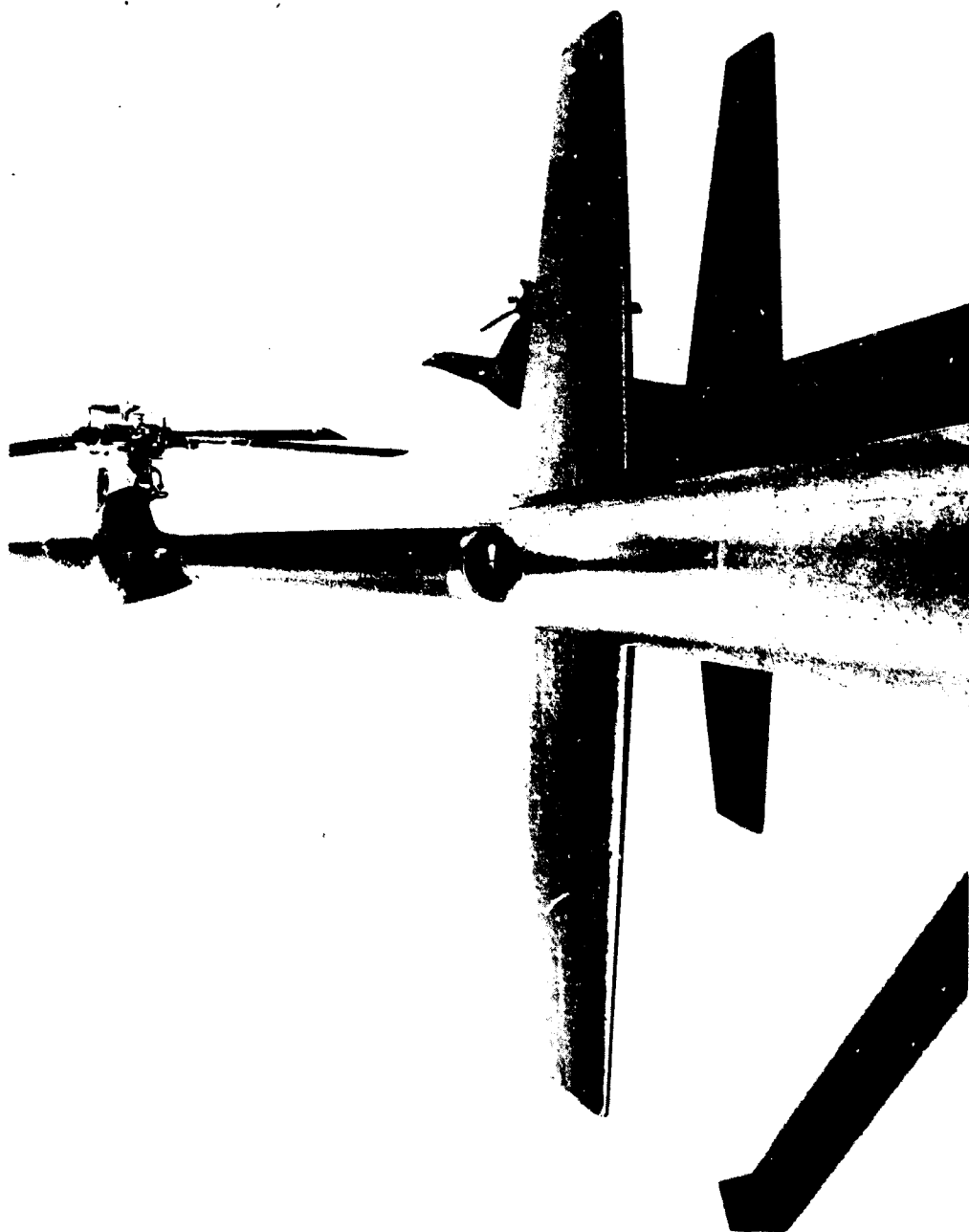
Figure 2. S-67 Stabilator in "Fly" Mode.

Figure 3. H-37 Ambulance in "Droop" Mode.
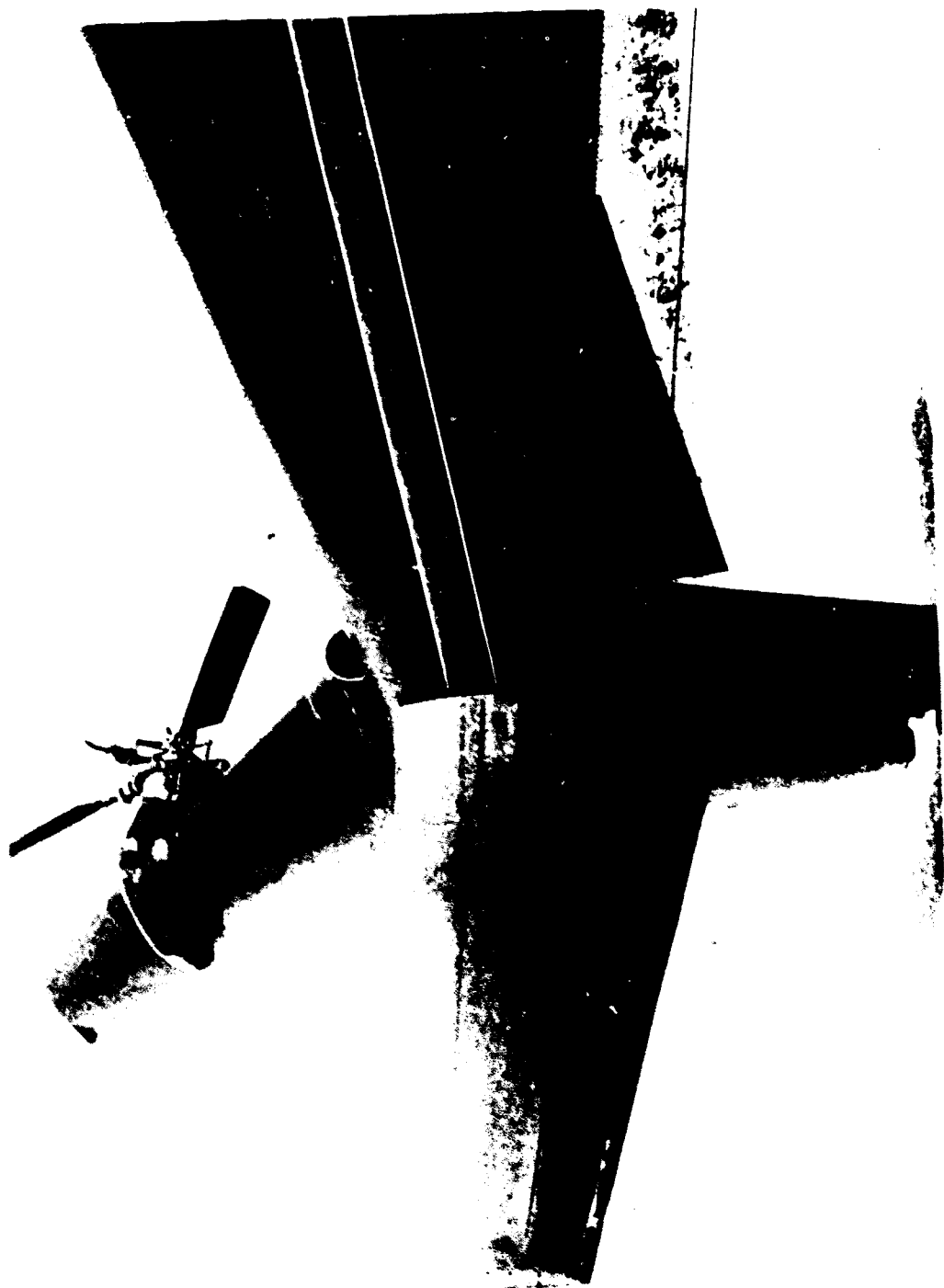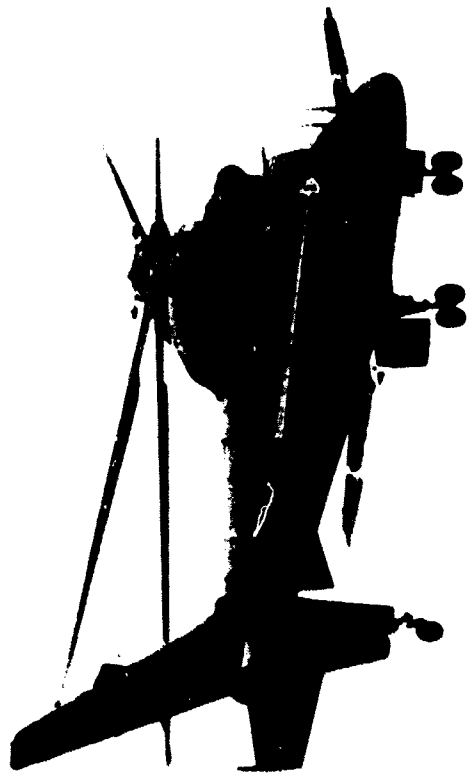
Figure 4. S-67 Aircraft in Hovering Flight With the Stabilator in the "Free" Mode.

LOCKING PIN ACTUATOR

BIAS ACTUATOR

TORQUE TUBE

F.S.
636.2

W.L.
149.8

STABILATOR

DROOP ACTUATOR

STABILATOR CONTROL SERVO

TAIL CONE

TO LONGITUDINAL
CYCLIC CONTROL
SYSTEM
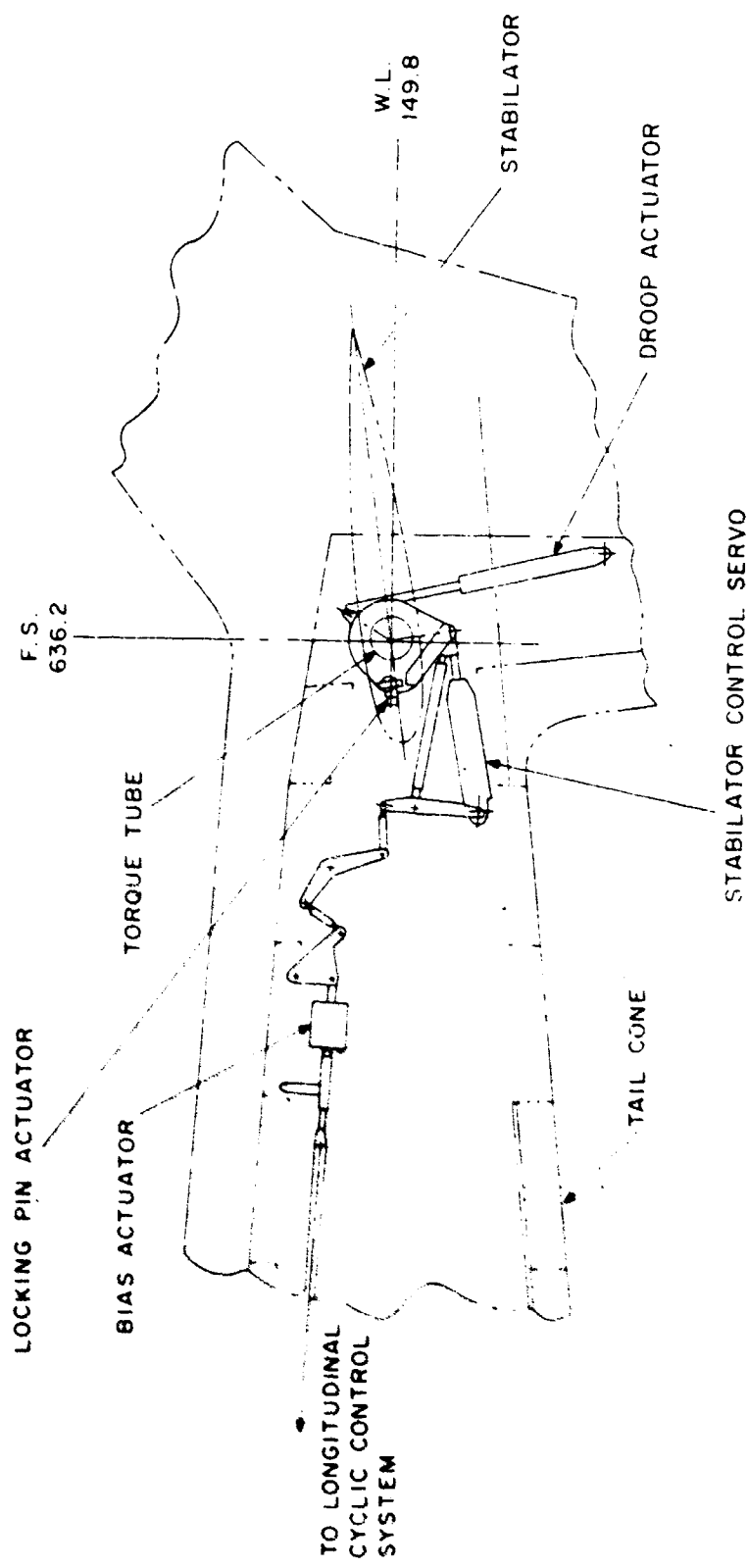
Figure 5. Stabilator Control System Schematic.
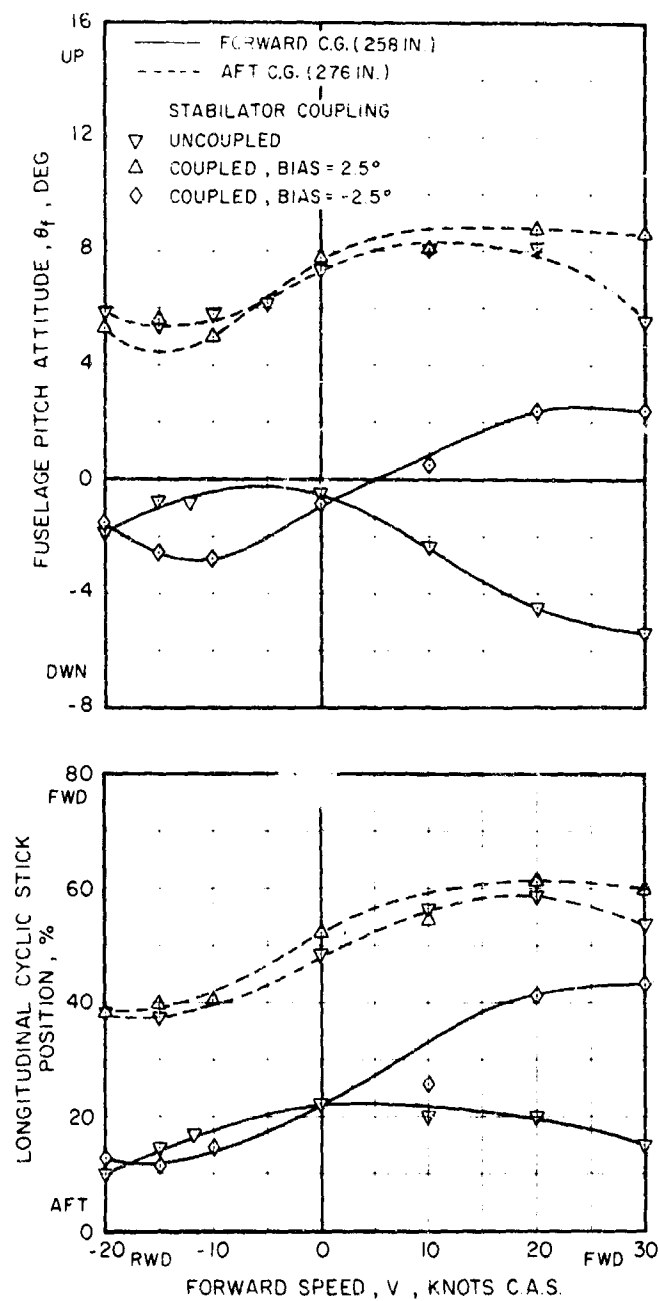
(a) LONGITUDINAL TRIM

Figure 6. Effect of Stabilator Coupling in Trimmed Hover and Low-Speed
Forward and Rearward Flight for Forward and Aft C.G. Positions;
G.W. = 17,300 Lb, Wheel Height = 30 Ft.
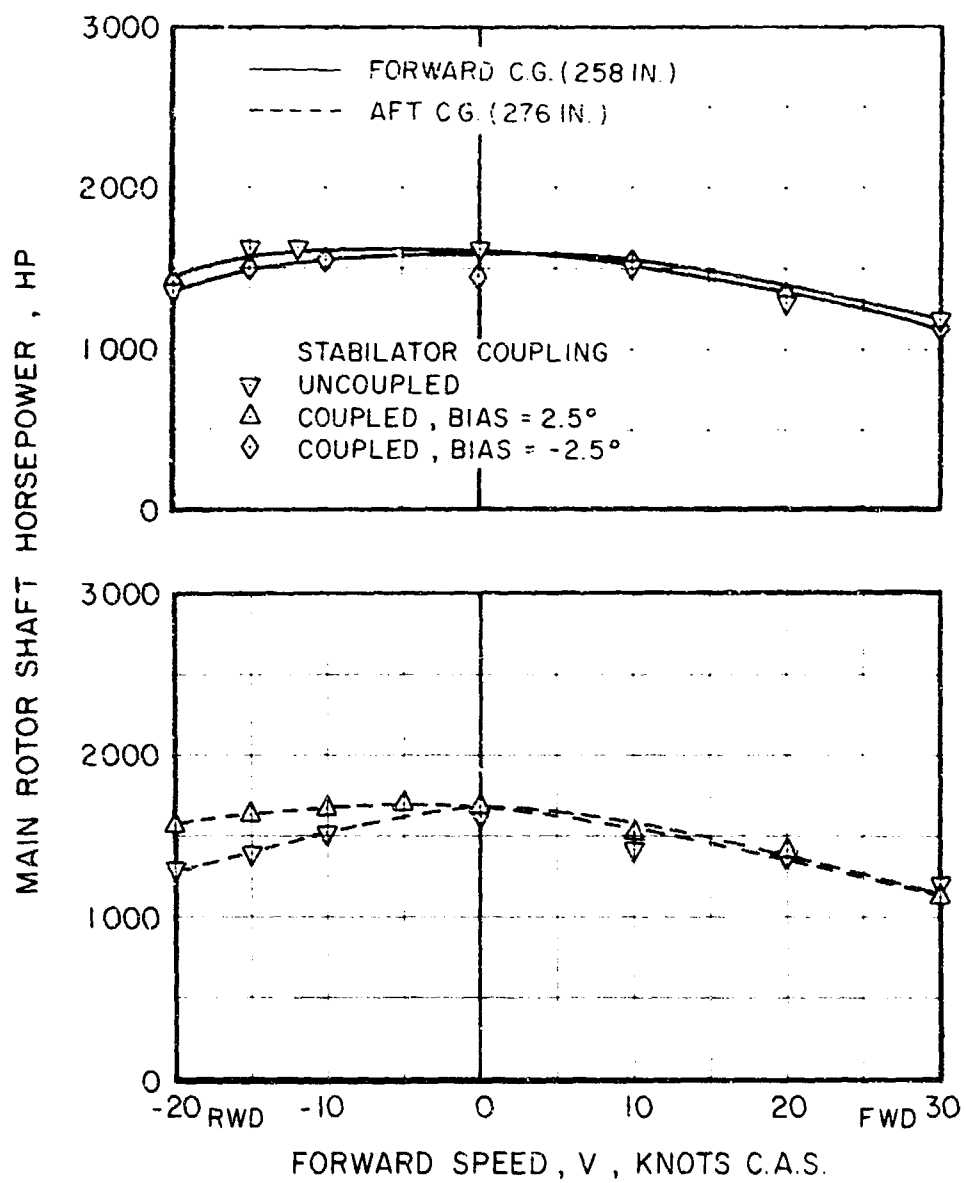
(b)   MAIN ROTOR SHAFT HORSEPOWER

Figure 6.   Concluded.

53
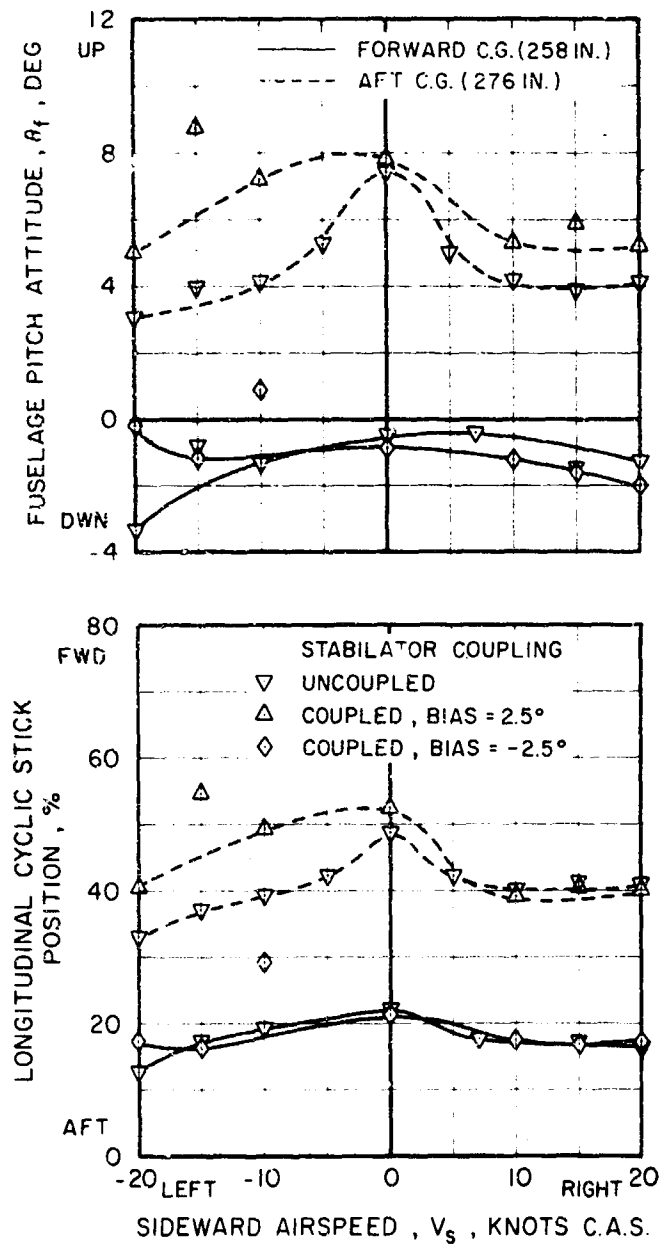
(a) LONGITUDINAL TRIM

Figure 7. Effect of Stabilator Coupling in Trimmed Hover and Sideward
Flight for Forward and Aft C.G. Positions; G.W. = 17,300 Lb,
Wheel Height = 30 Ft.

34
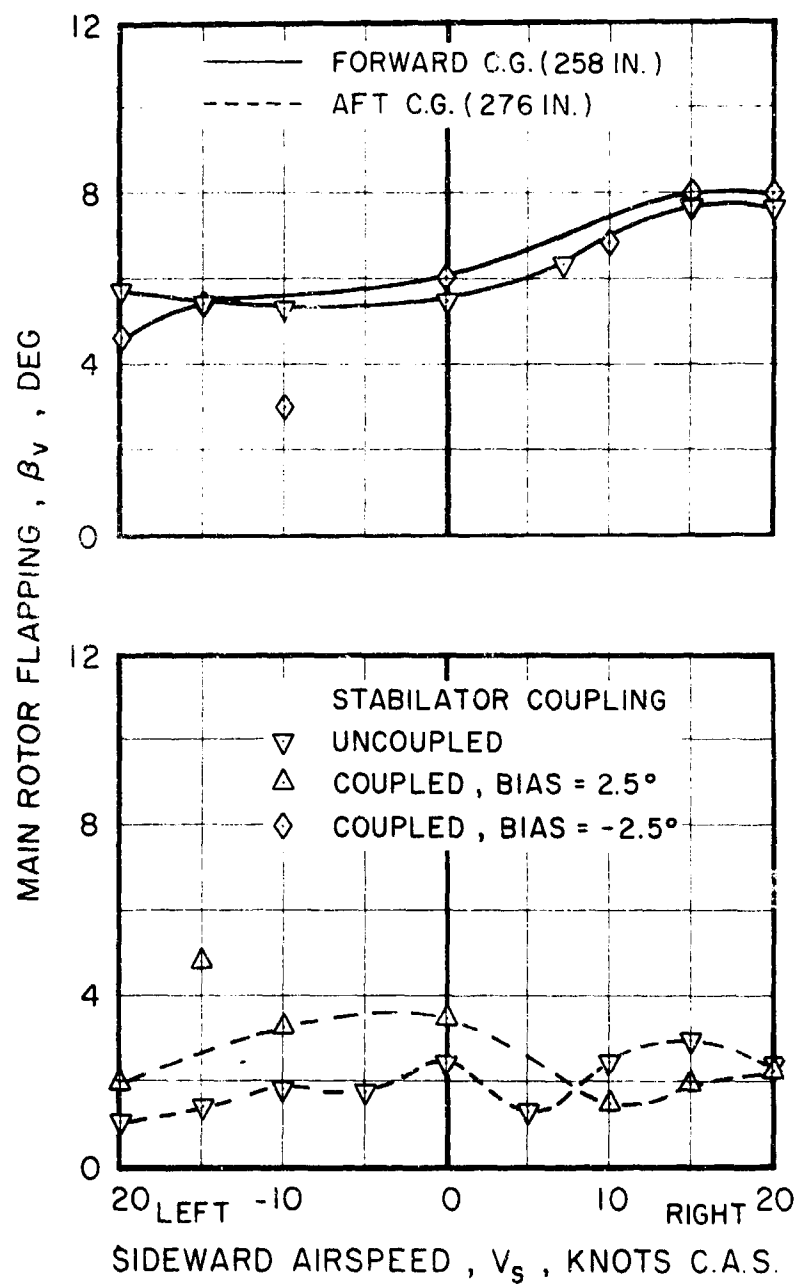
(b) MAIN ROTOR FLAPPING

Figure 7. Continued.
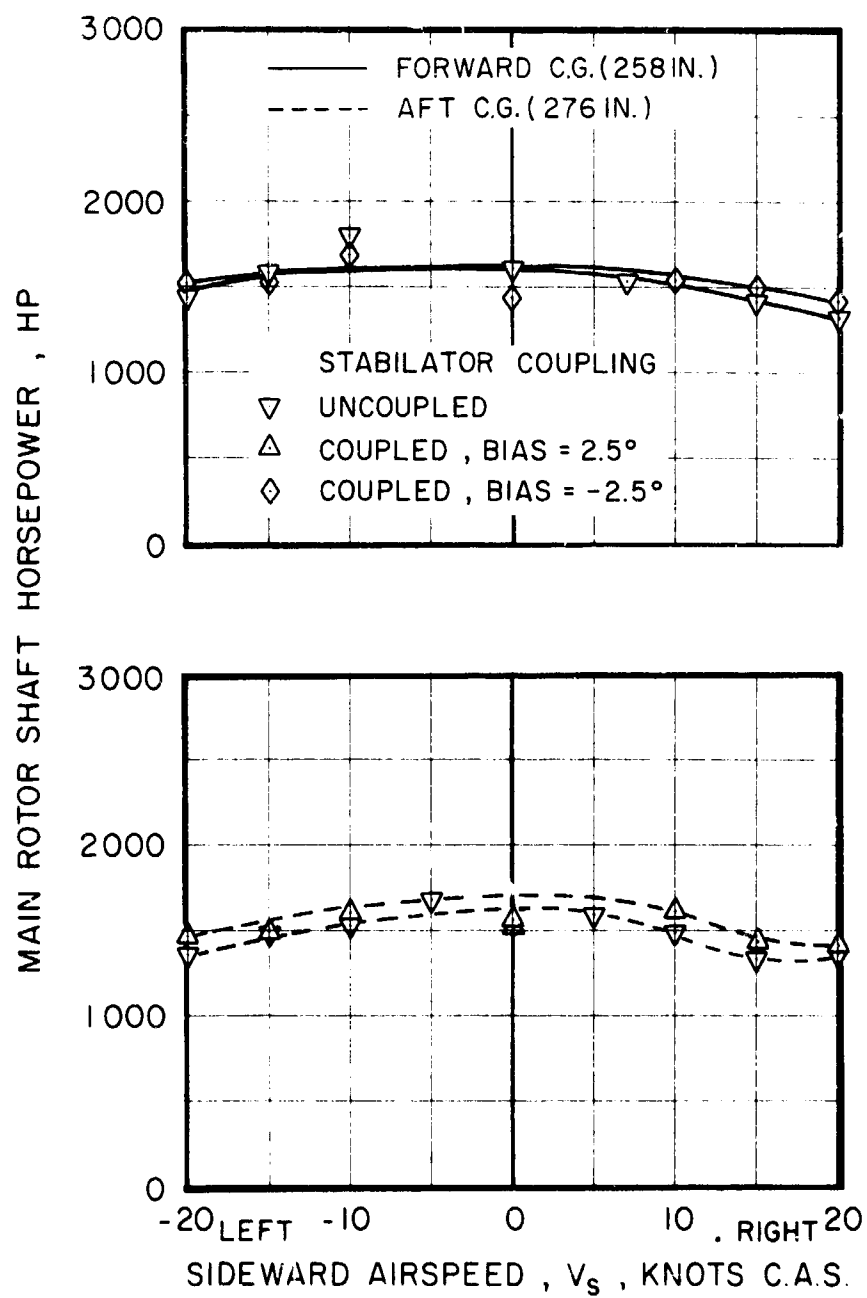
(c)   MAIN ROTOR SHAFT HORSEPOWER , HP

Figure 7.   Continued.

36
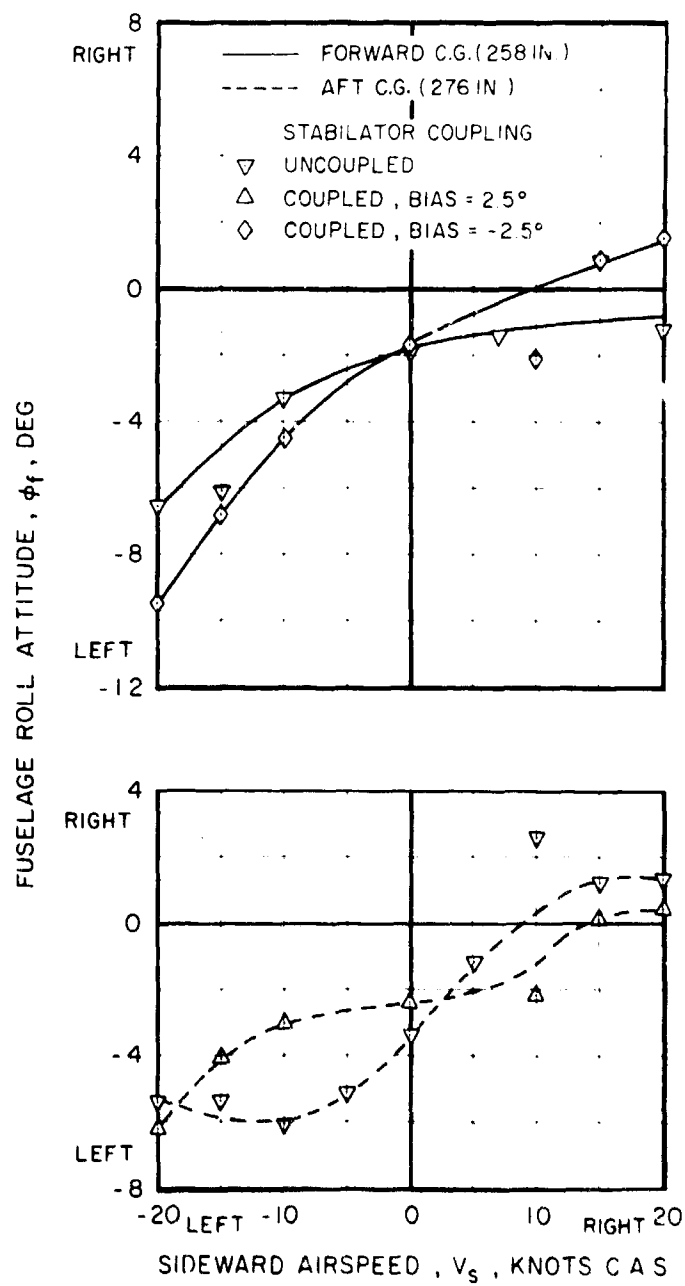
(d) FUSELAGE ROLL ATTITUDE

Figure 7. Continued.
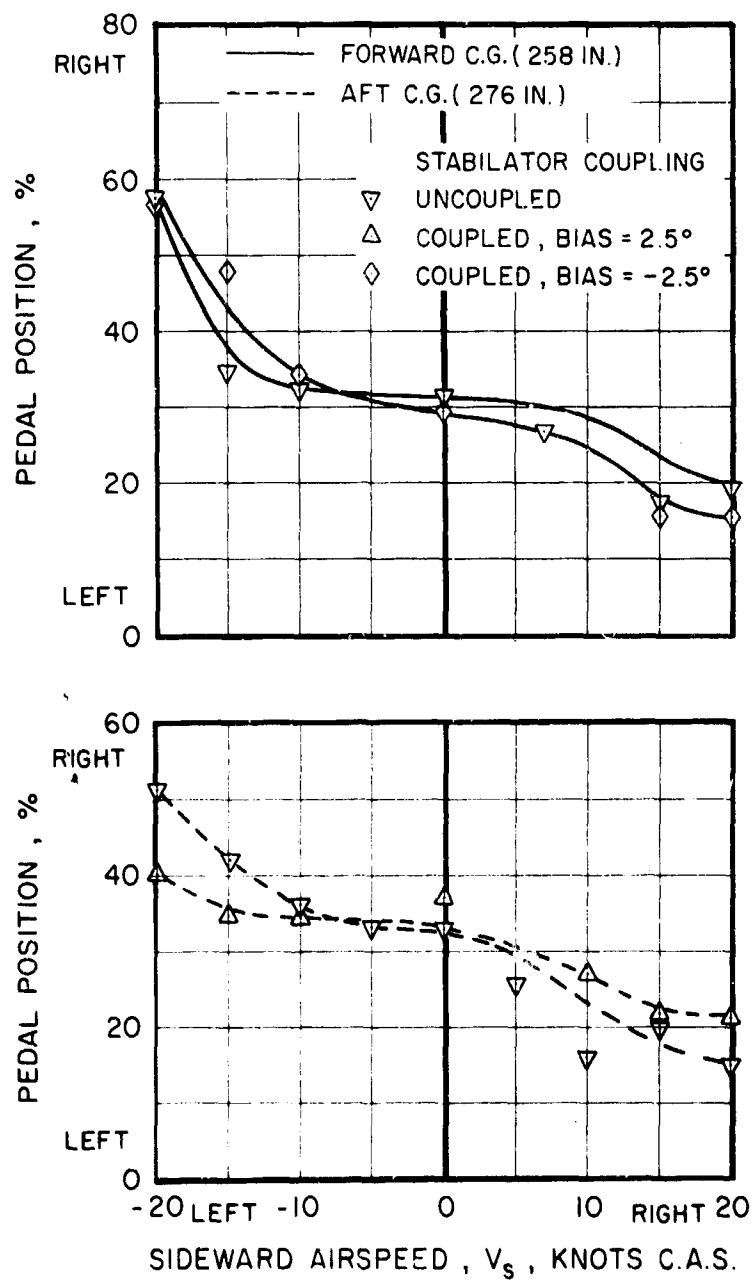
(e)   PEDAL POSITION

Figure 7.   Concluded.
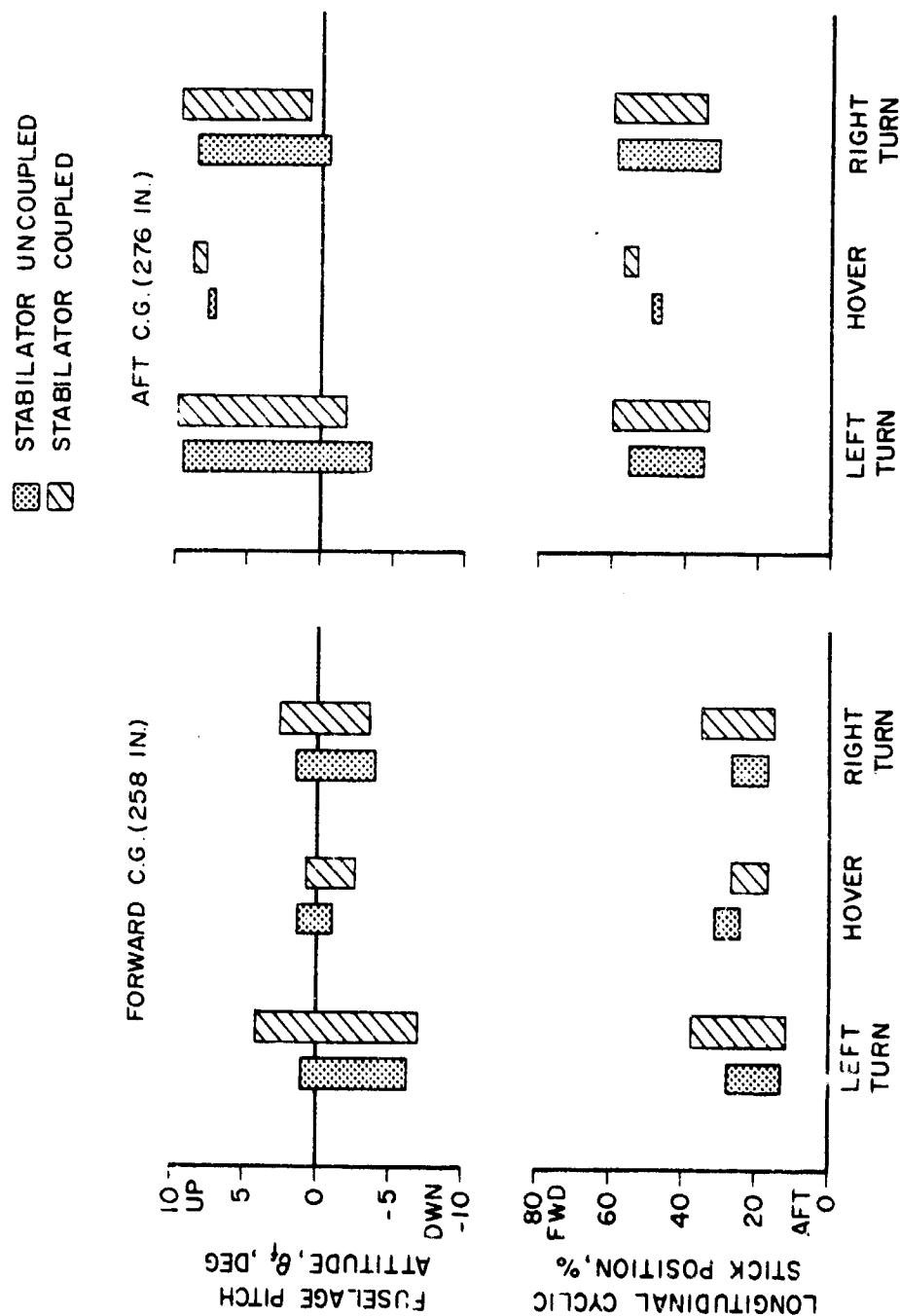
Figure 8. Effect of Stabilator Coupling on Total Longitudinal Control Displacement and Aircraft Pitch Attitude Change During Hover and Hovering Turns for Forward and Aft C.G. Positions; G.W. = 17,300 Lb.
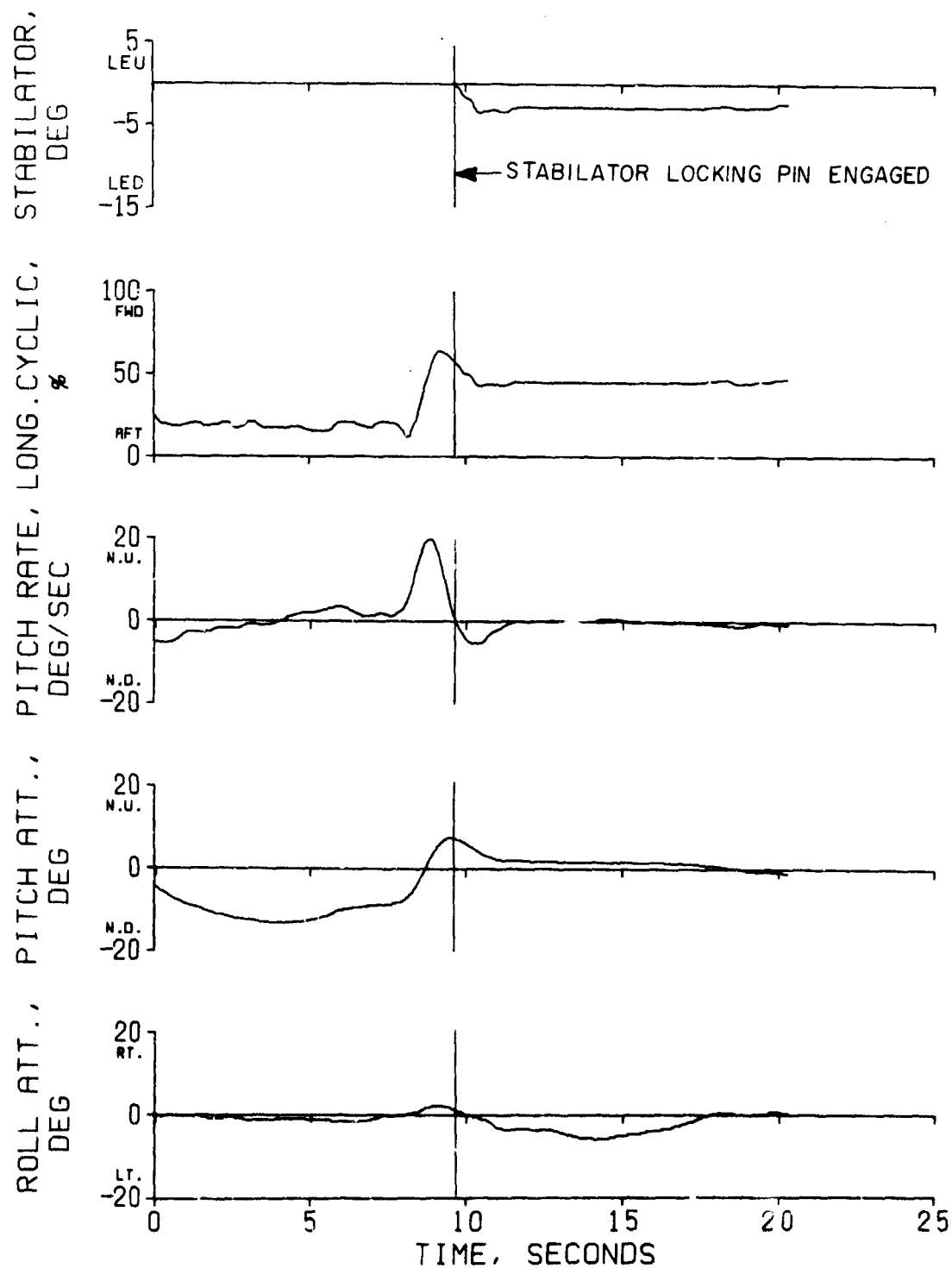
Figure 9. Time History of Aircraft and Control Motions During a Takeoff Transition Flight With Stabilator Coupled to "Fly" Mode at 40 Kn; Bias = -2.5 Deg, G.W. = 17,300 Lb. C.G. = 258 In.
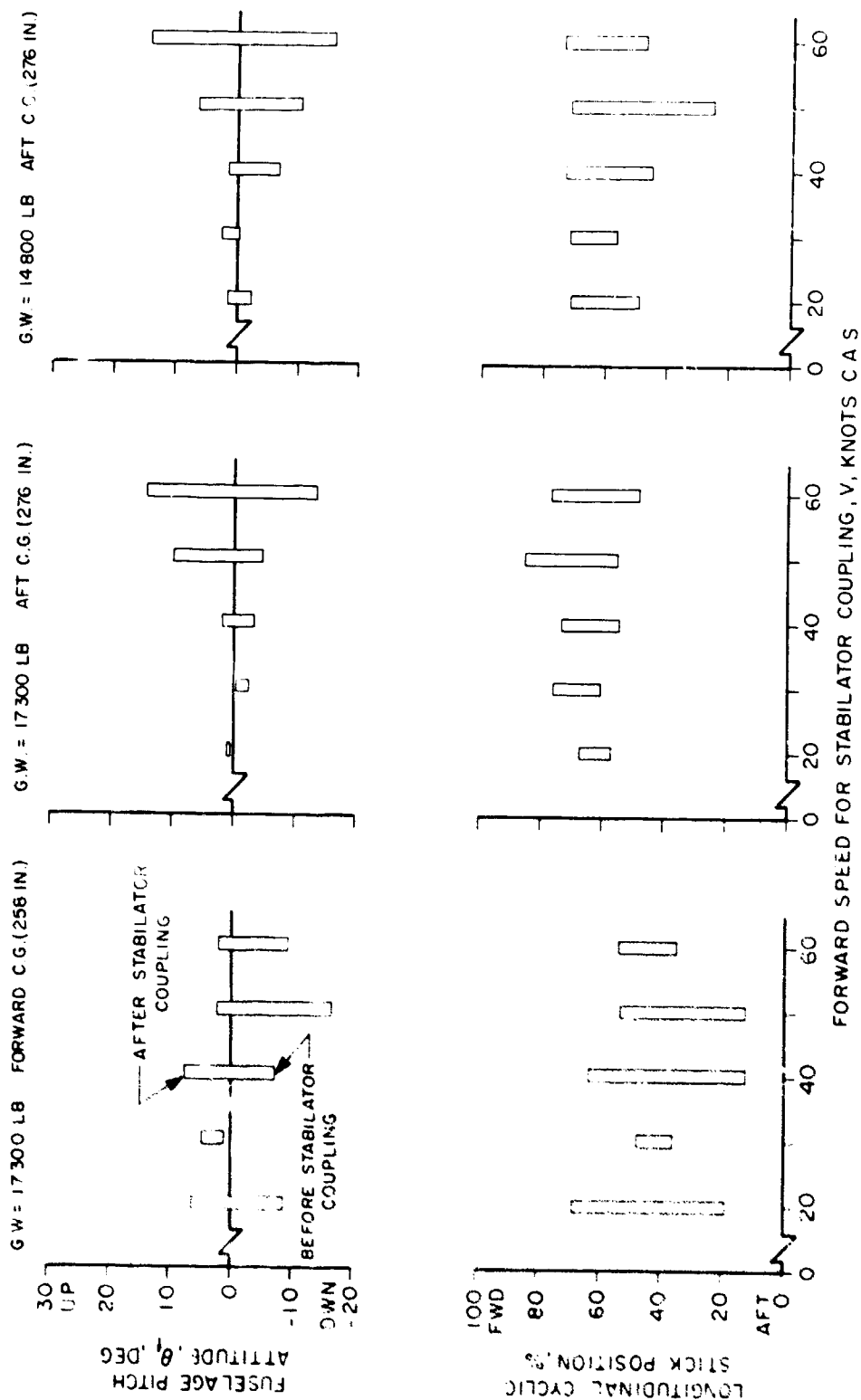
Figure 22. Longitudinal Control Motion and Aircraft Pitch Attitude Response Following Stabilator Coupling to "Fly" Mode in Accelerating Takeoff Transition Flight for Several G.W. and C.G. Loading Conditions.
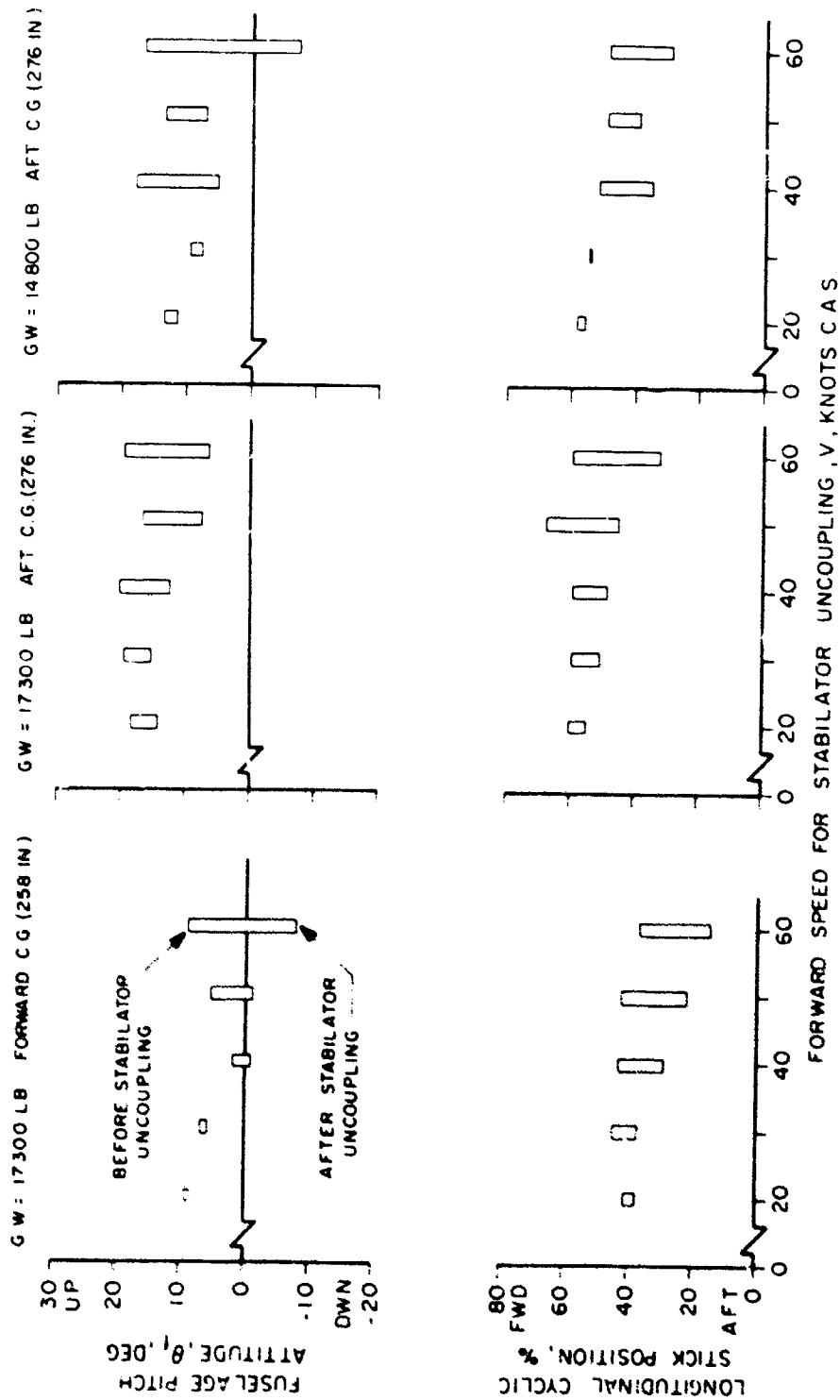
Figure 11. Longitudinal Control Motion and Aircraft Pitch Attitude Response Following Stabilator Uncoupling to "Free" Mode in Decelerating Approach Transition Flight for Several G.W. and C.G. Loading Conditions.
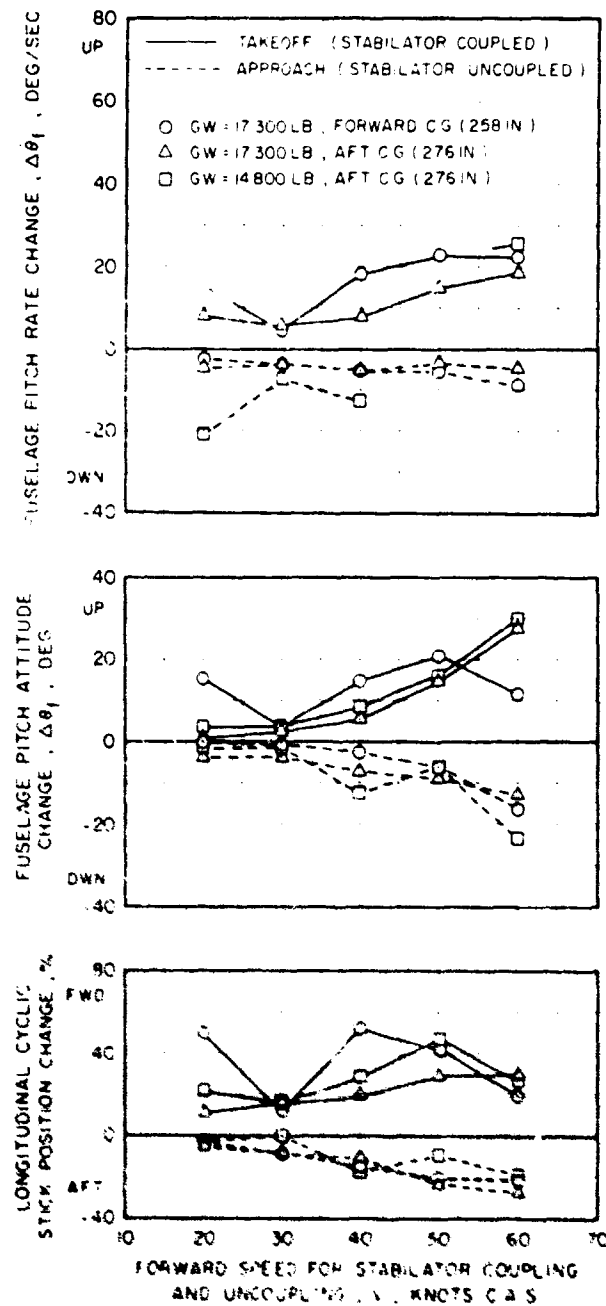
Figure 12. Comparison of Longitudinal Control Displacement and Aircraft Pitching Motion Following Stabilator Coupling and Uncoupling in Transition Flight for Several G.W. and C.G. Loading Conditions.
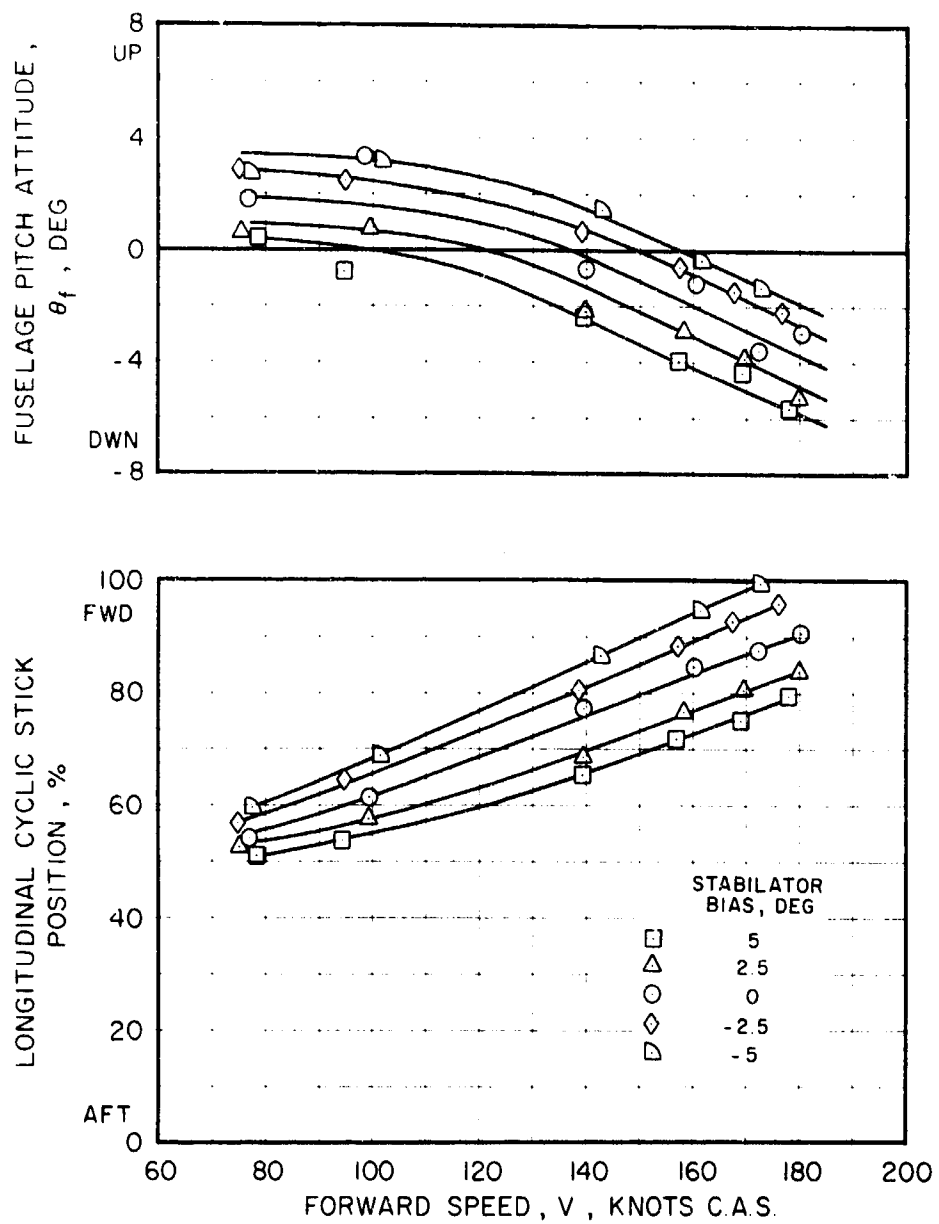
(a) LONGITUDINAL TRIM

Figure 13. Effect of Stabilator Bias Setting on Aircraft Trim
Characteristics in High-Speed Level Flight;
G.W. = 17,300 Lb, C.G. = 258 In.

44
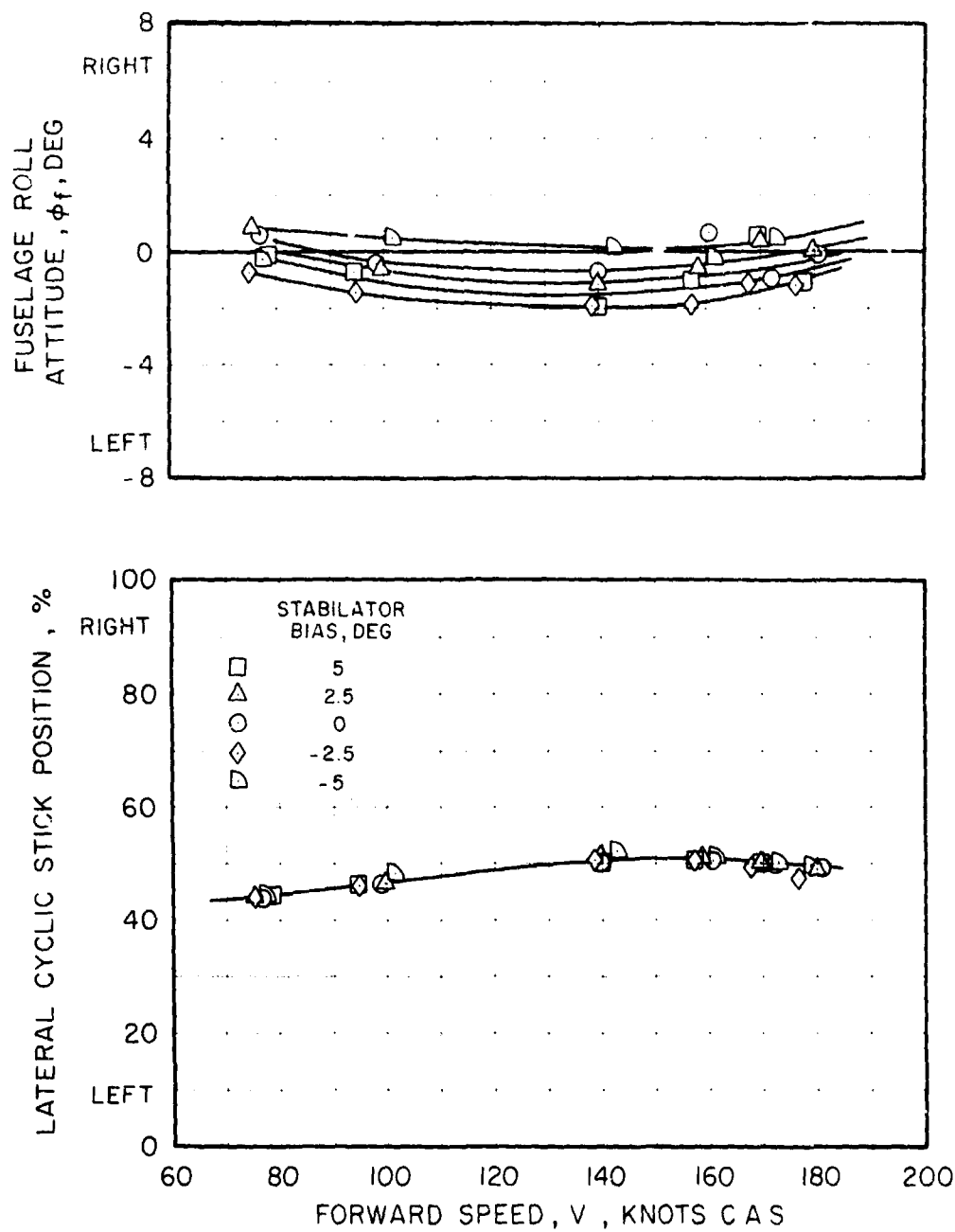
(b) LATERAL TRIM

Figure 15. Continued.

45
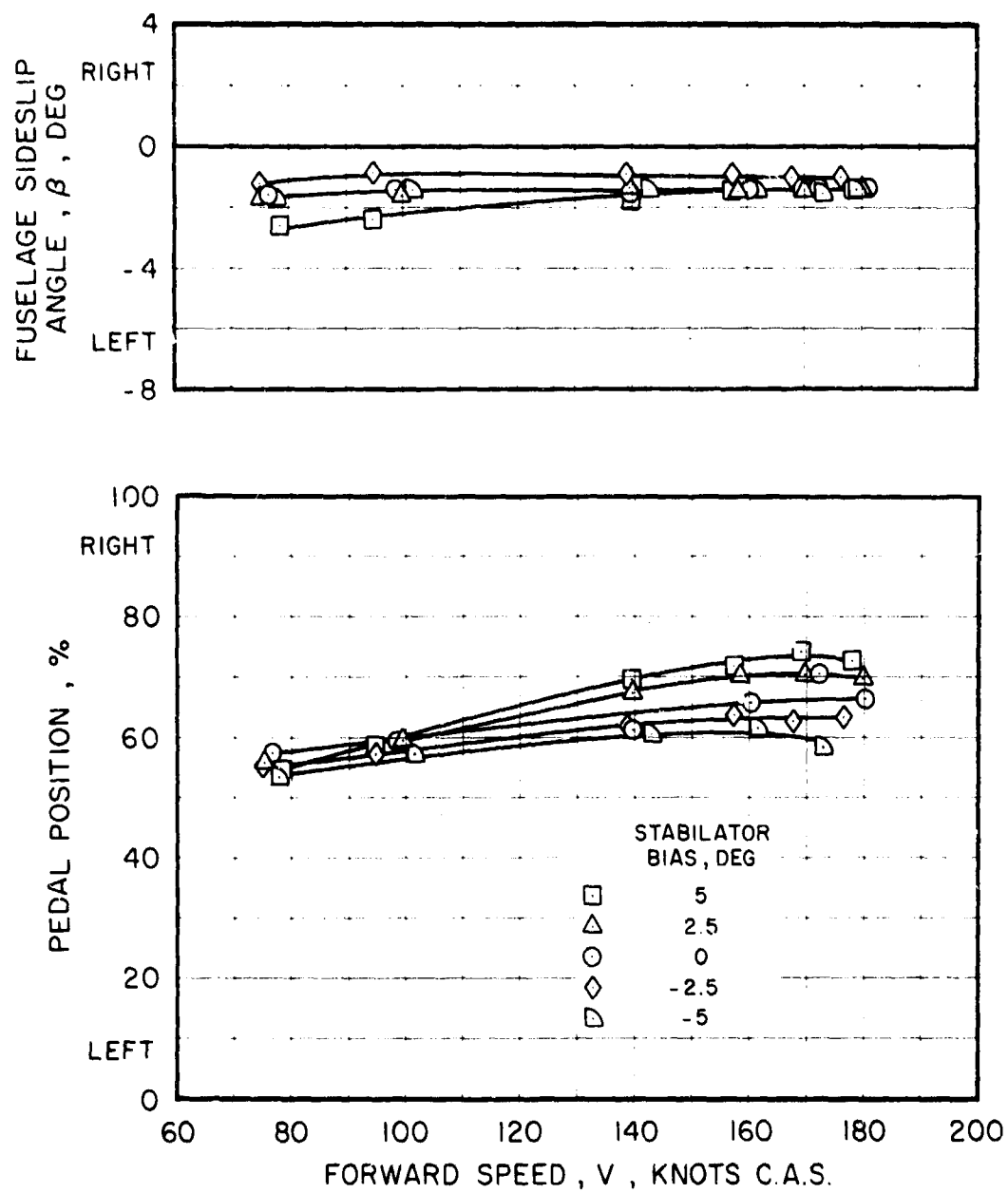
(c) YAW TRIM

Figure 13. Continued.
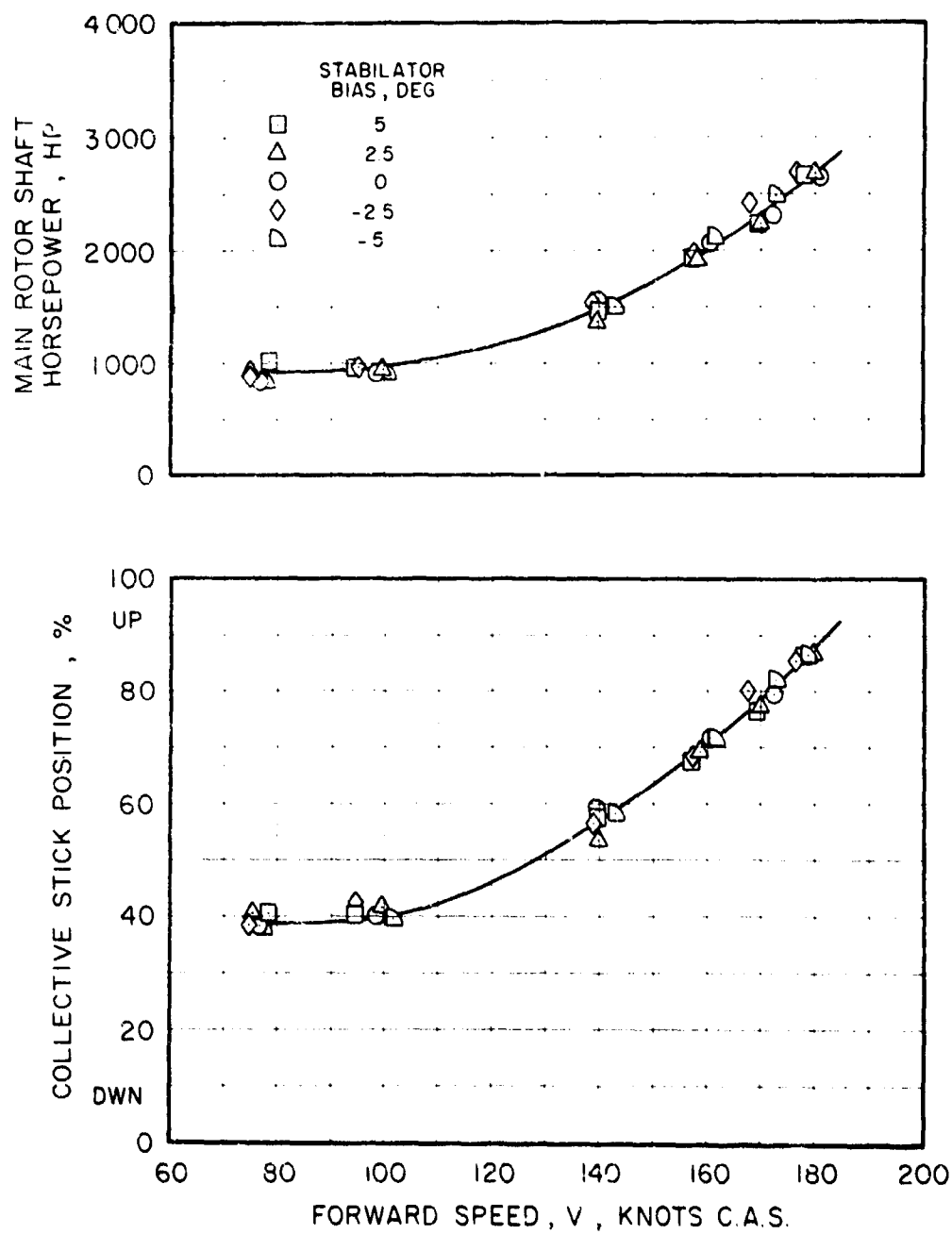
(d) VERTICAL TRIM

Figure 13. Continued.

47
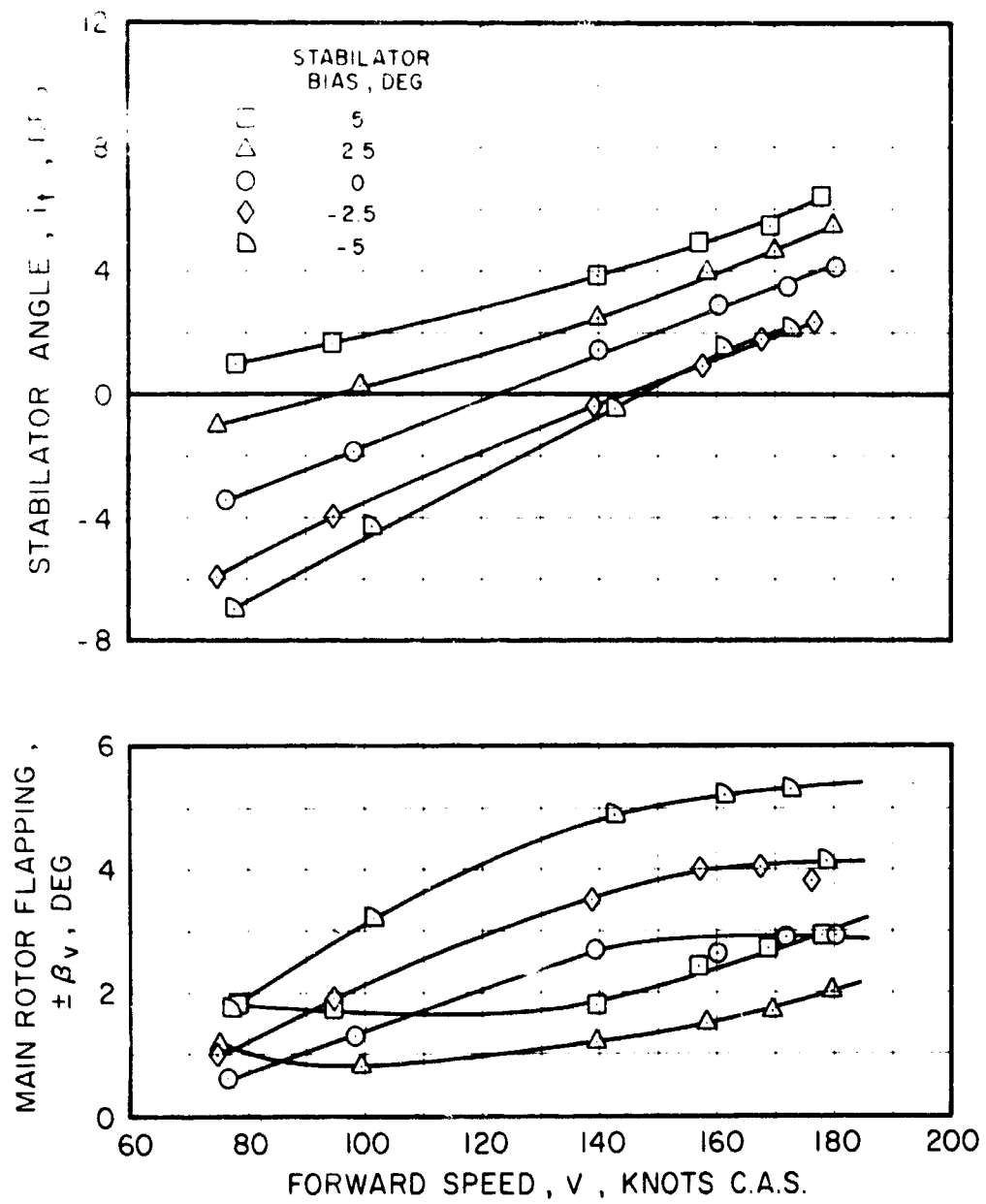
(e) MAIN ROTOR FLAPPING AND STABILATOR ANGLE

Figure 13. Continued.

48
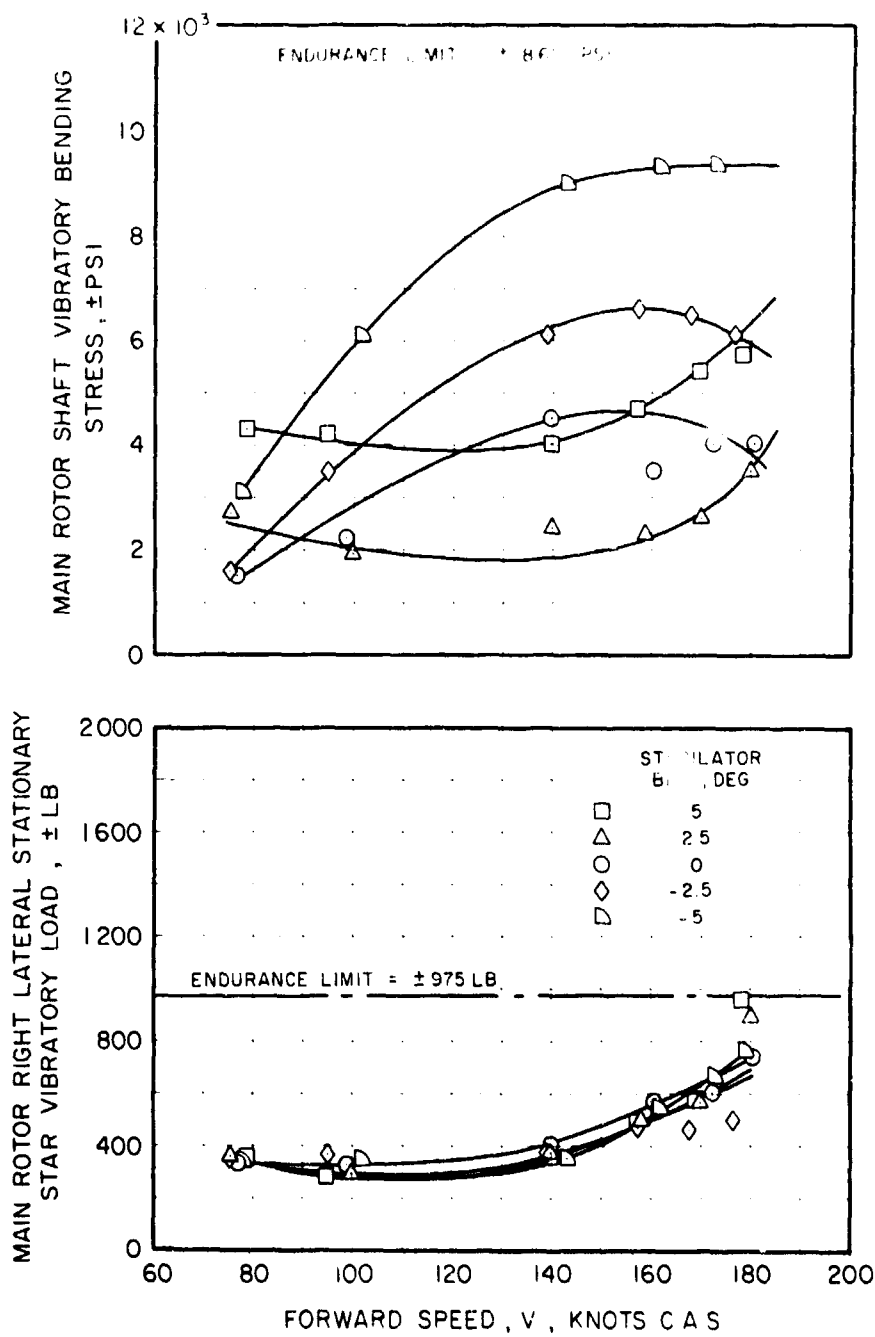
(f) MAIN ROTOR STAR LOAD AND SHAFT BENDING STRESS

Figure 13. Continued.
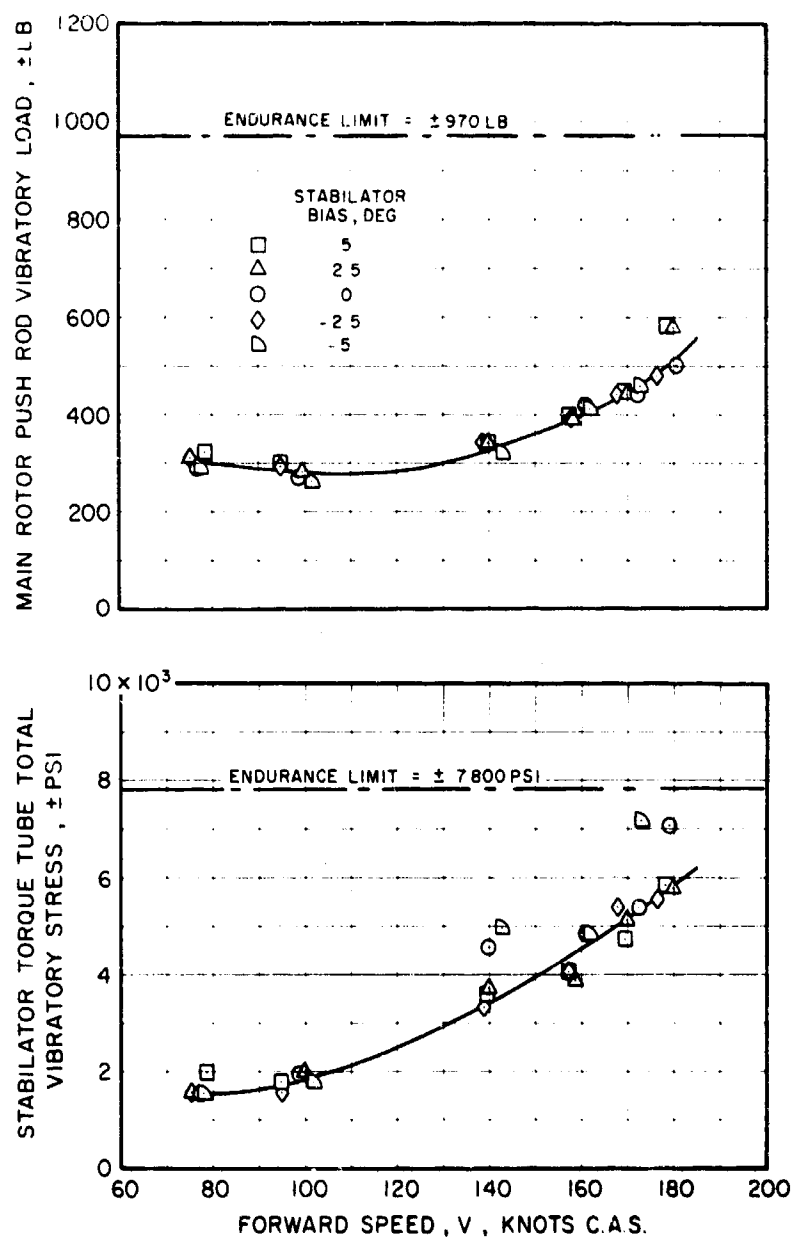
(g) STABILATOR TORQUE TUBE STRESS AND
MAIN ROTOR PUSH-ROD LOAD

Figure 13.   Concluded.

50
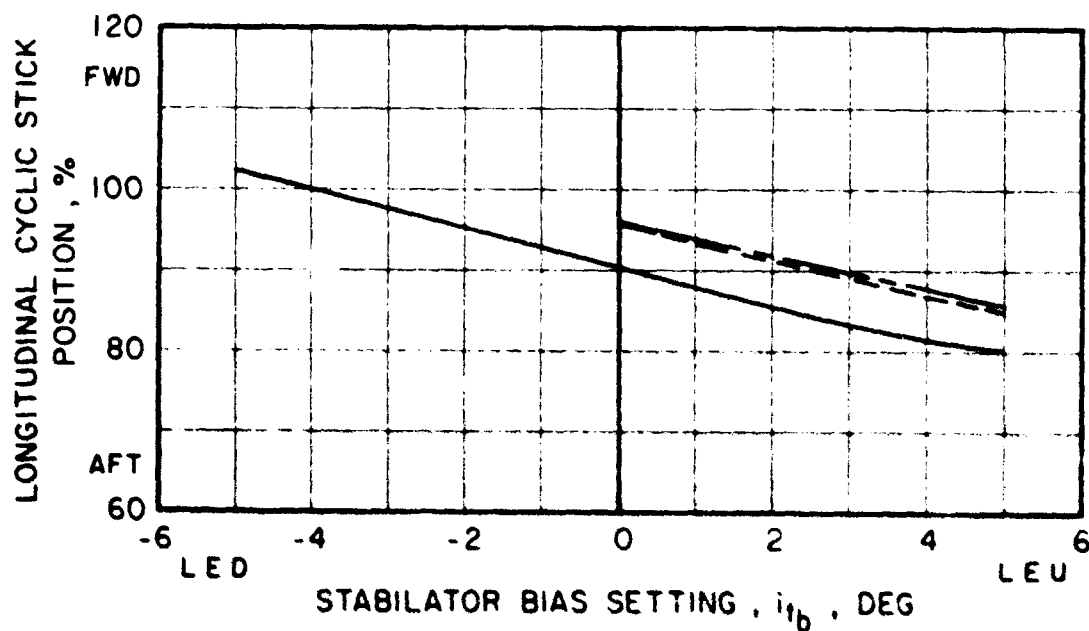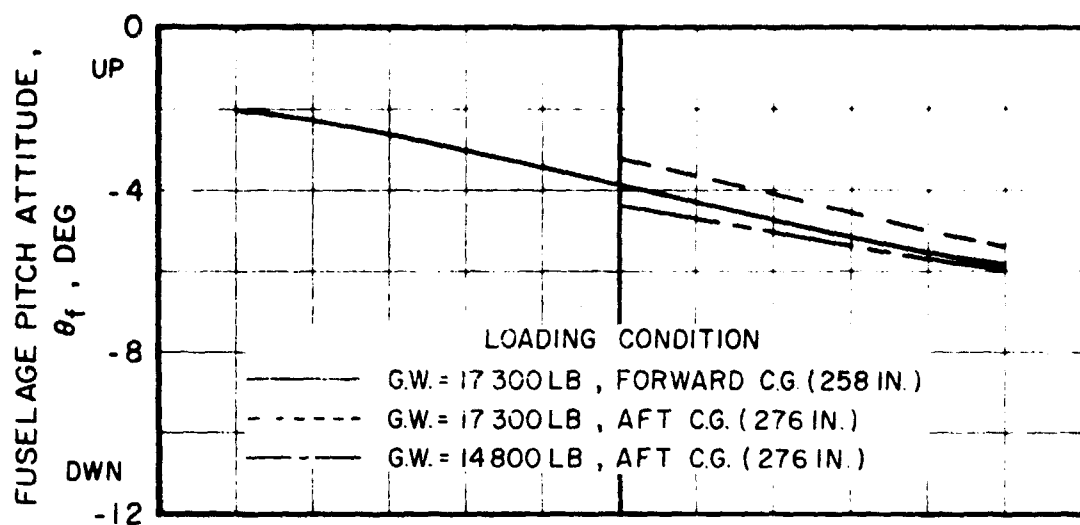
Figure 14. Variation of Longitudinal Cyclic Stick Position and Fuselage Pitch Attitude With Stabilator Bias Setting at 180-Knot Trimmed Level Flight for Several G.W. and C.G. Loading Conditions.
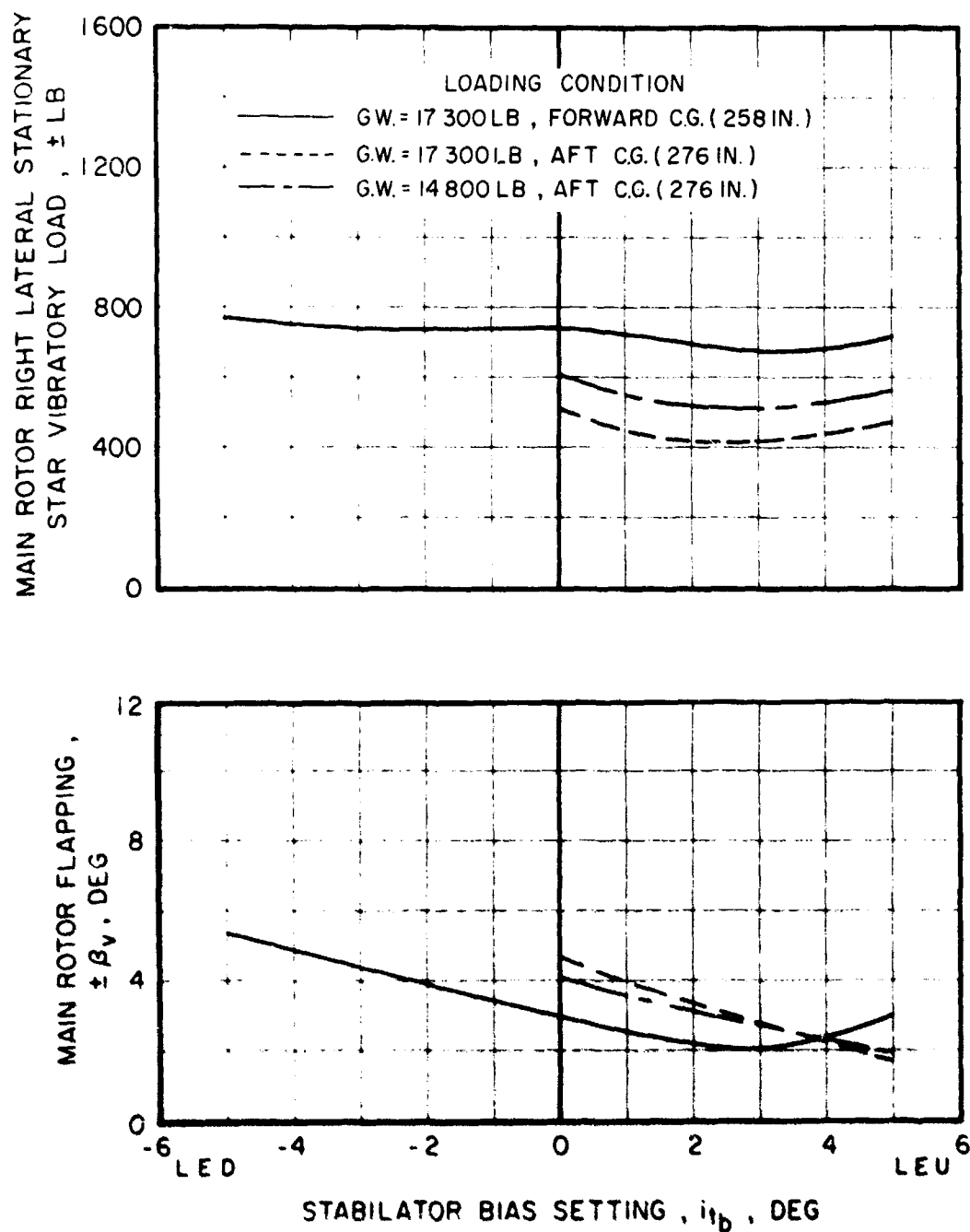
Figure 15. Variation of Main Rotor Flapping and Stationary Star Load With Stabilator Bias Setting at 180-Knot Trimmed Level Flight for Several G.W. and C.G. Loading Conditions.

52
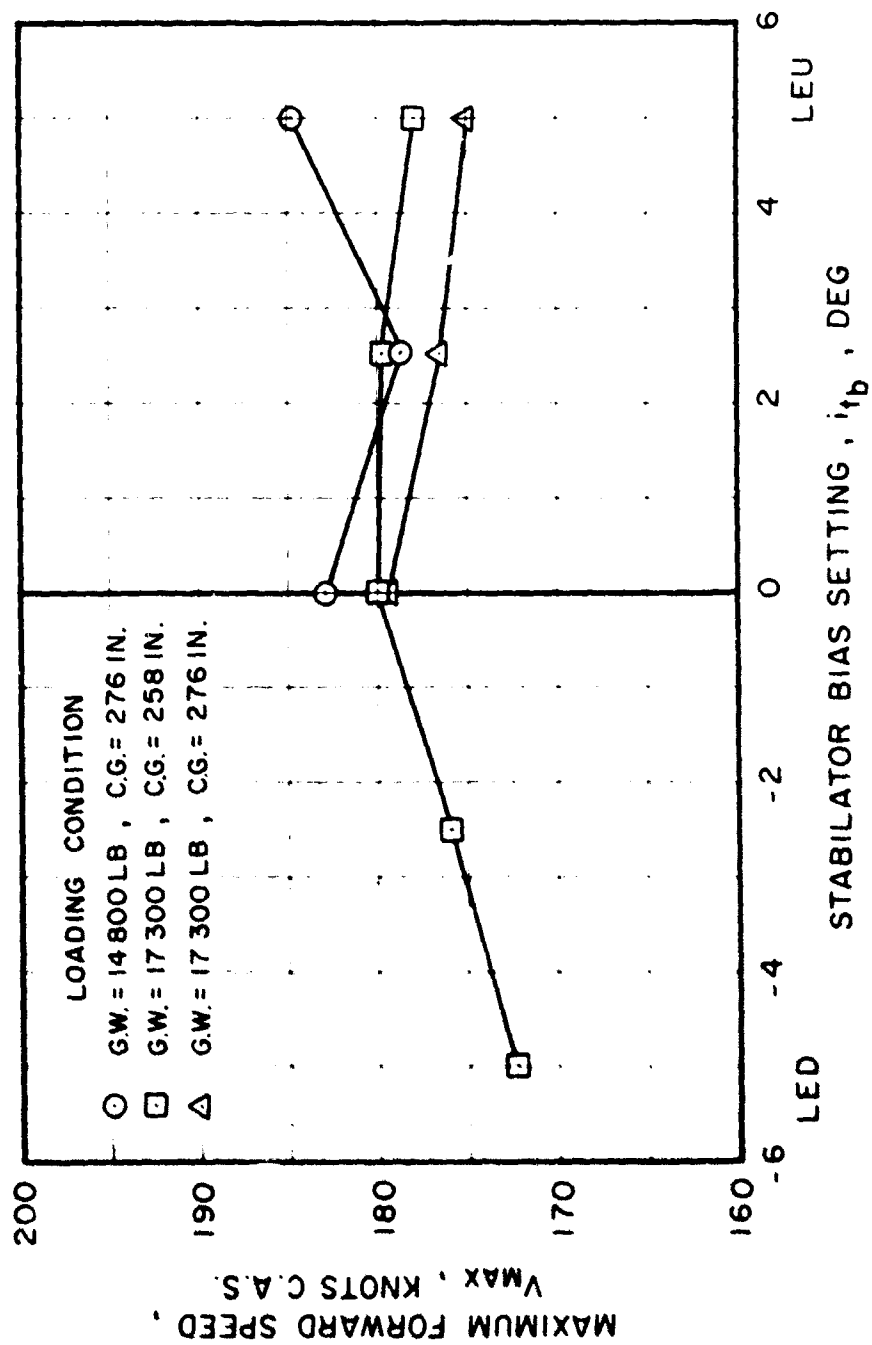
Figure 22. Variation of Maximum Level Flight Forward Speed With Stabilator Bias Setting for Several G.W. and C.G. Loading Conditions.
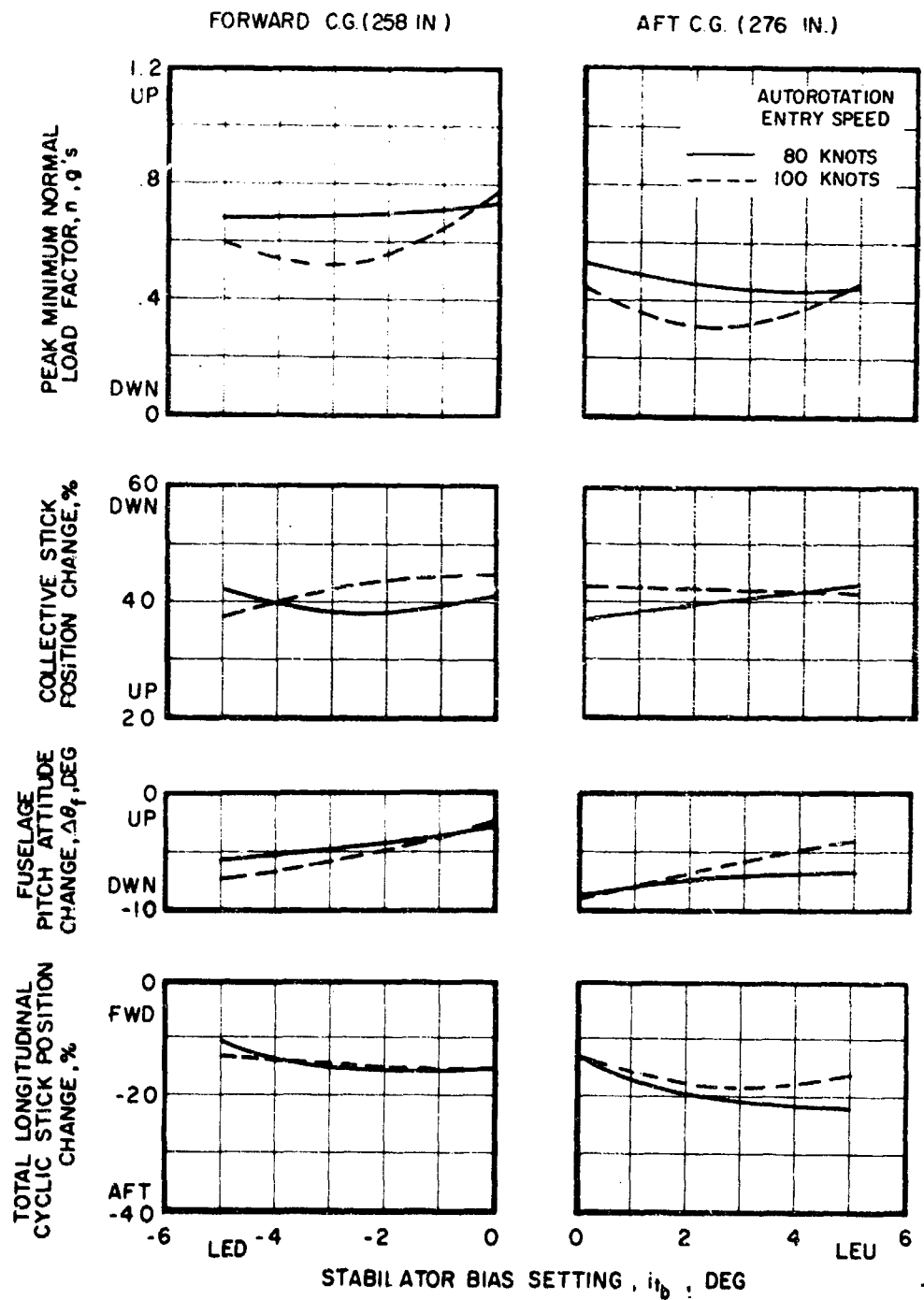
Figure 17. Longitudinal Control Displacement and Aircraft Pitching Motion During 80- and 100-Knot Autorotation Entry Maneuvers for Forward and Aft C.G. Positions, G.W. = 17,300 Lb.
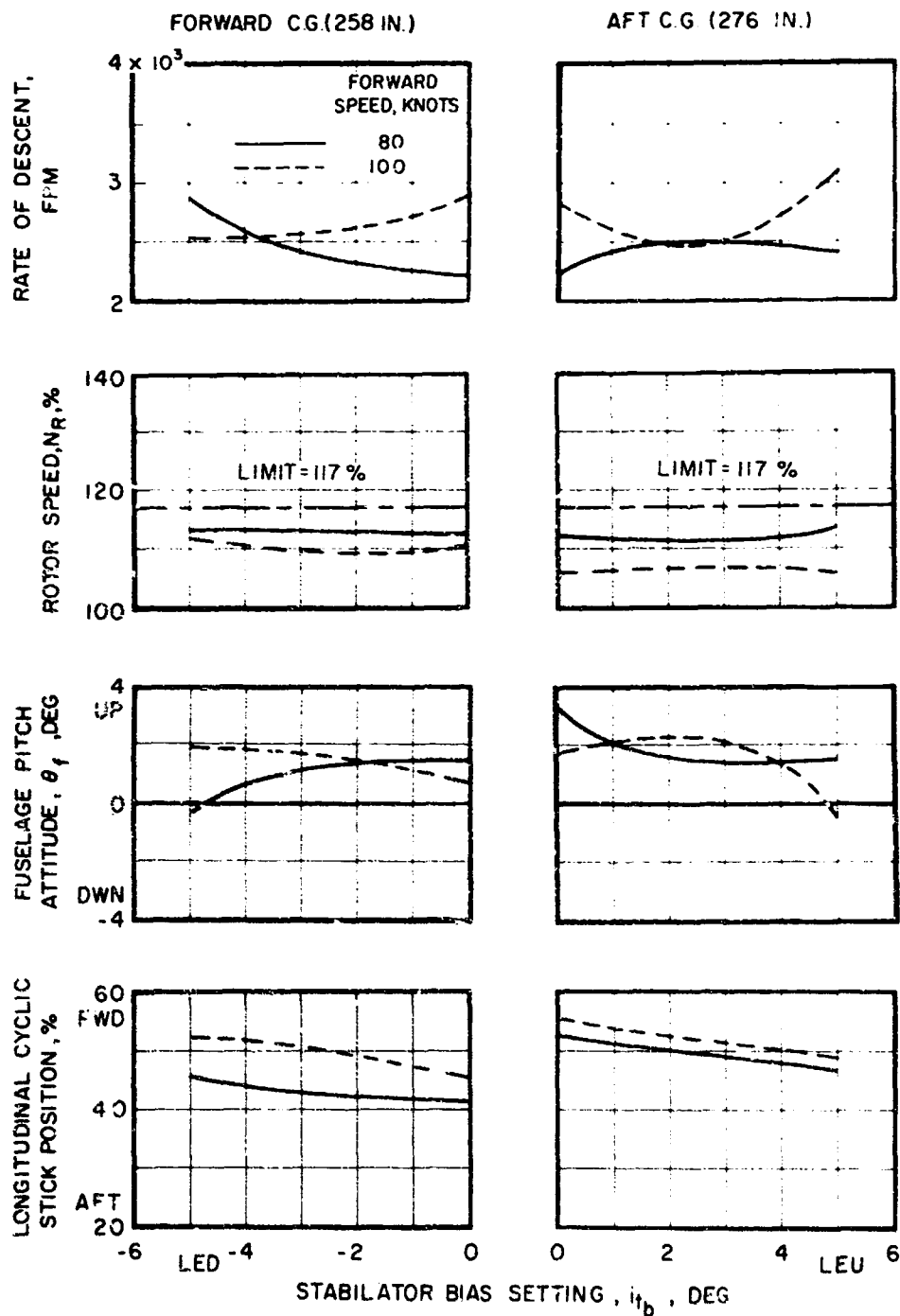
Figure 18. Variation of Longitudinal Trim Characteristics With
Stabilator Bias Setting During 80- and 100-Knot
Steady Autorotation for Forward and Aft C.G.
Positions; G.W. = 17,300 Lb.
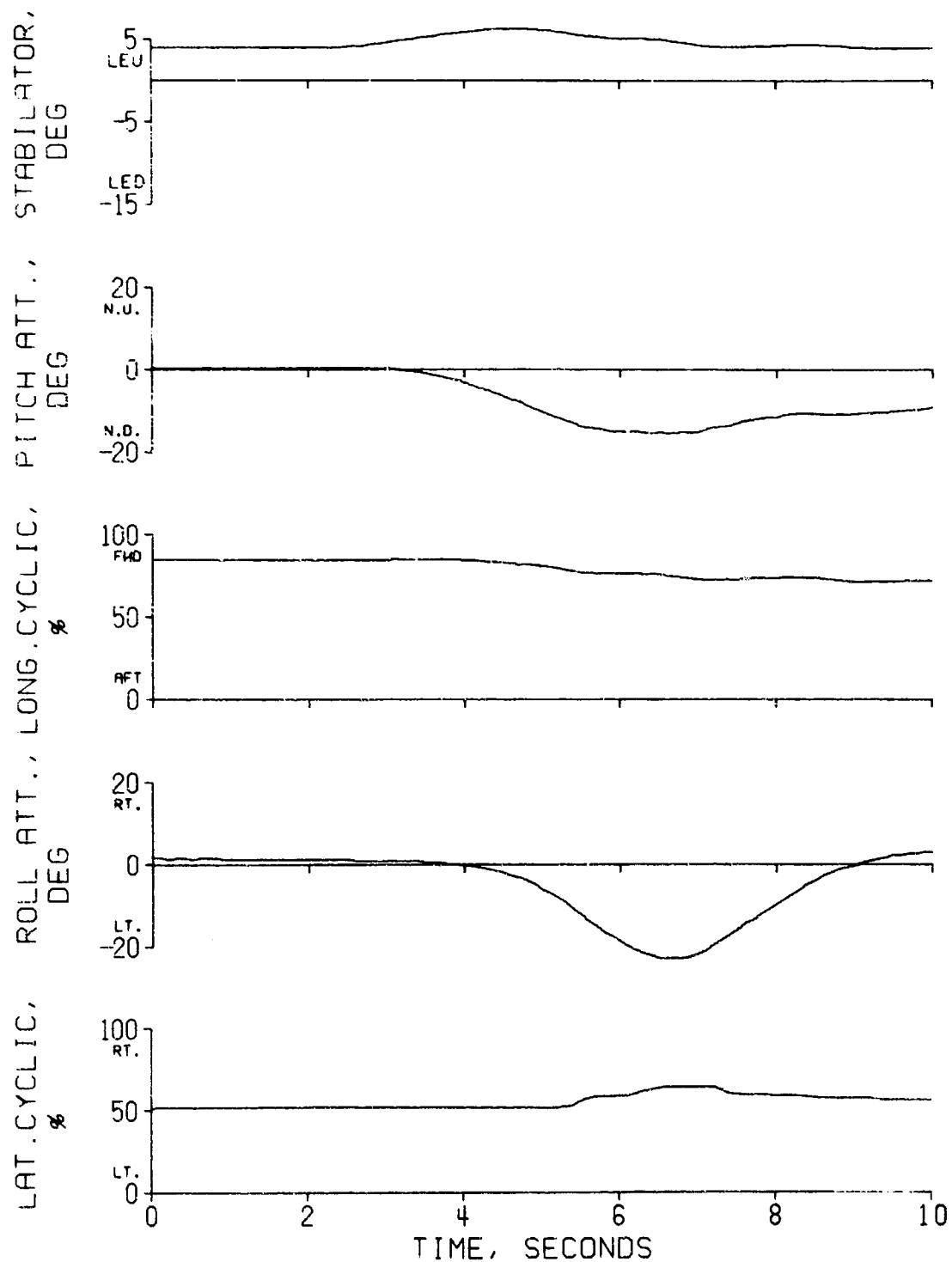
Figure 19. Time History of Aircraft and Control Motions Following
a Leading-Edge-Up Bias Actuator Hardover at 160 Knots;
Trim Bias = 2.5 Deg, Bias Hardover = 2.5,
G.W. = 17,300 Lb, C.G. = 276 In.

56
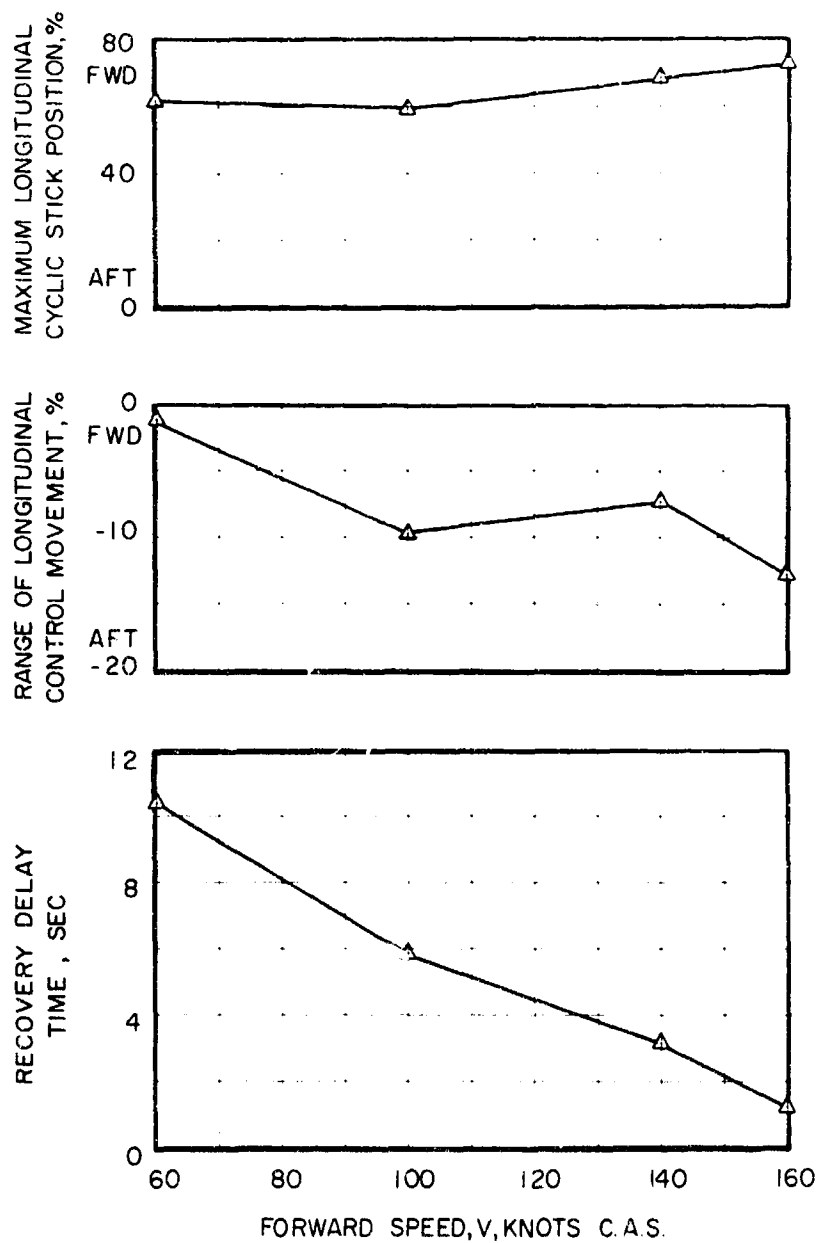
Figure 20. Recovery Delay Time, Longitudinal Control Motion,
            and Aircraft Response Following a Leading-Edge-Up
            Bias Actuator Hardover in High-Speed Flight;
            Trim Bias = 2.5 Deg, Bias Hardover = 2.5 Deg,
            G.W. = 17,300 Lb, C.G. = 276 In.
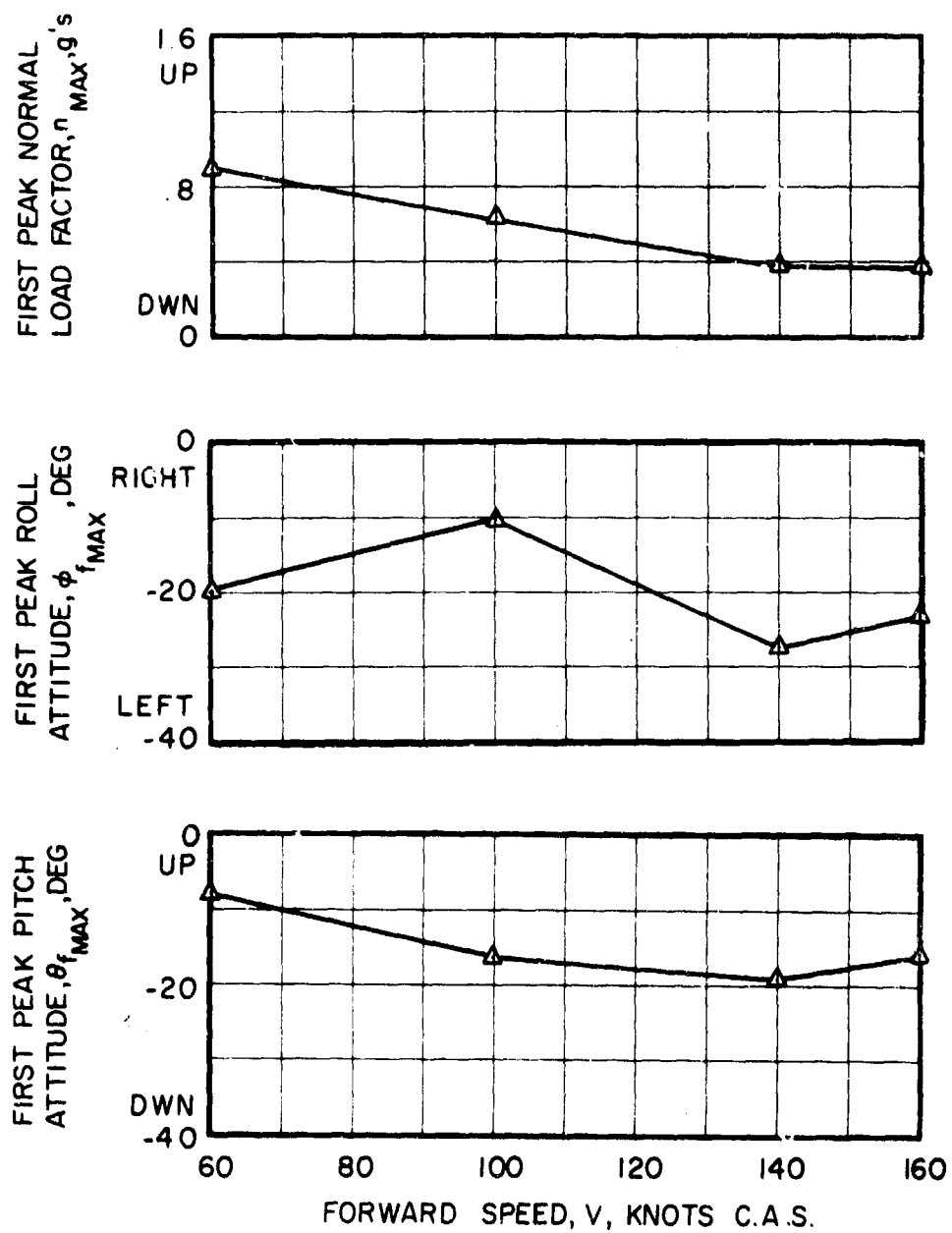
(b)

Figure 20. Concluded.
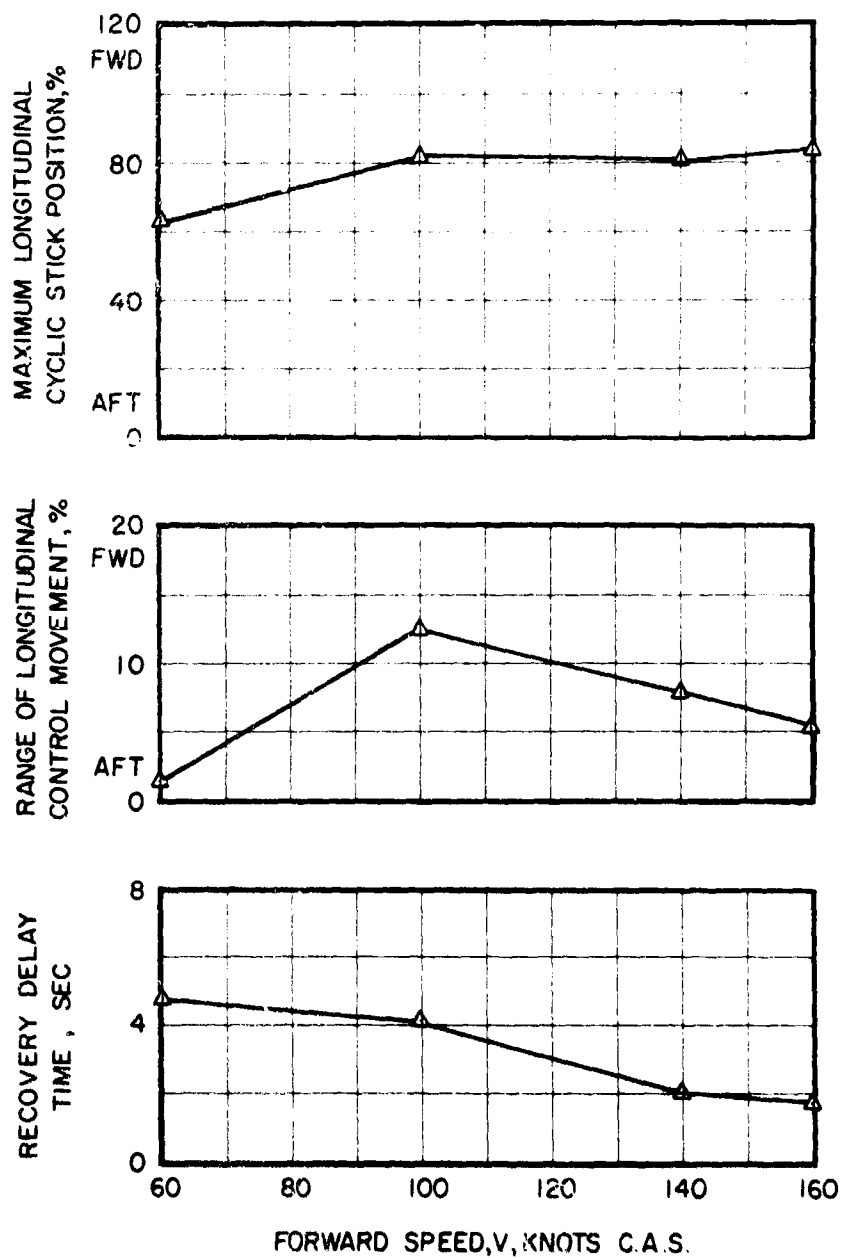
Figure 21. Recovery Delay Time, Longitudinal Control Motion, and Aircraft Response Following a Leading-Edge-Down Bias Actuator Hardover in High-Speed Flight; Trim Bias = 2.5 Deg, Bias Hardover = -7.5 Deg, G.W. = 17,300 Lb, C.G. = 276 In.
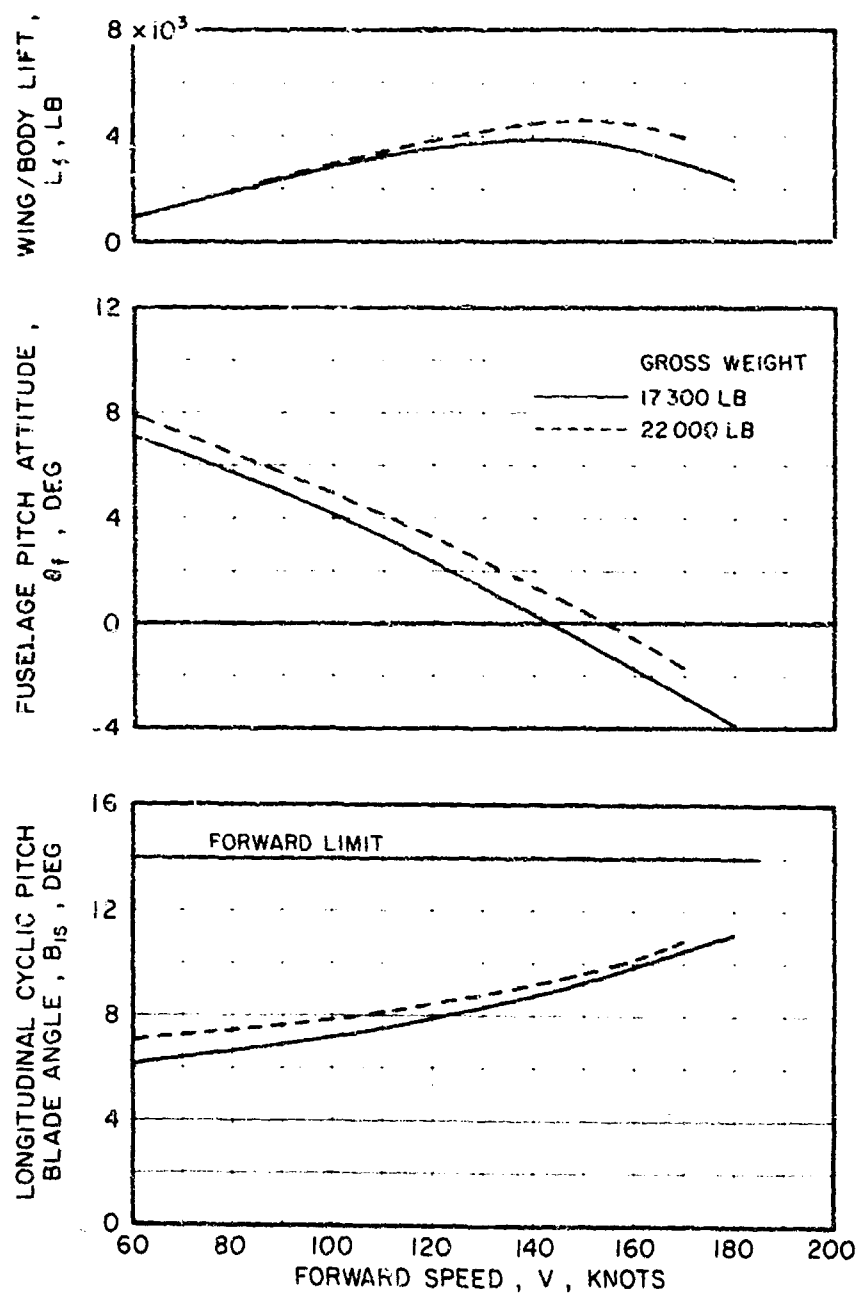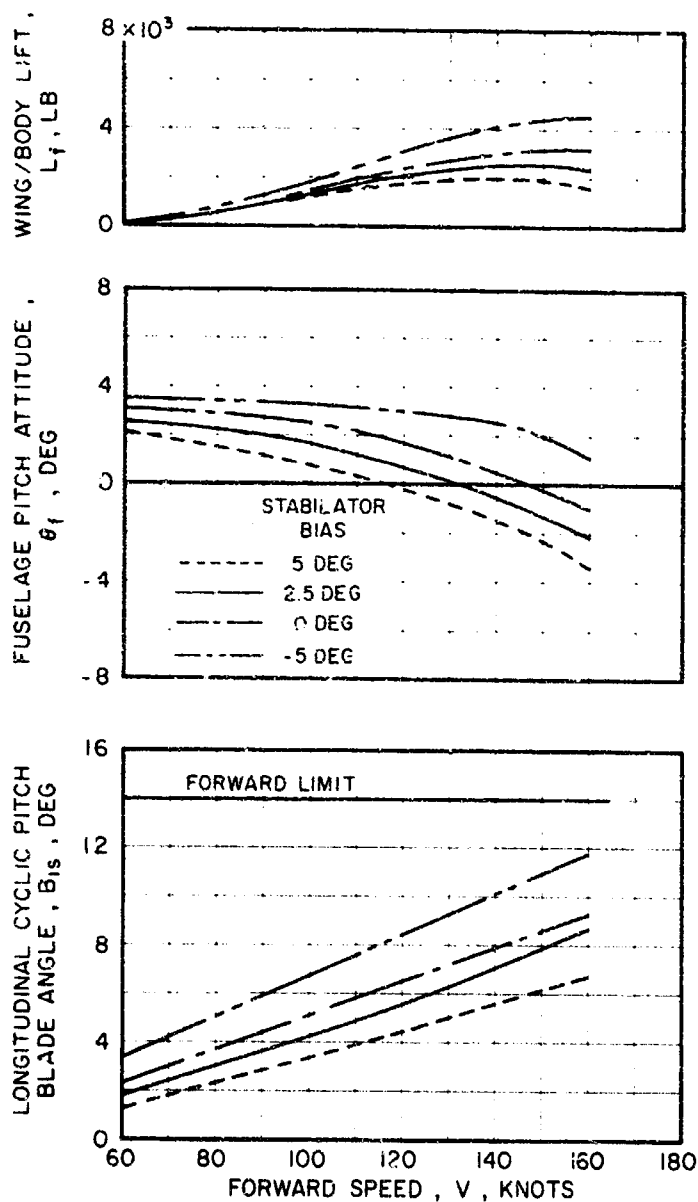
59
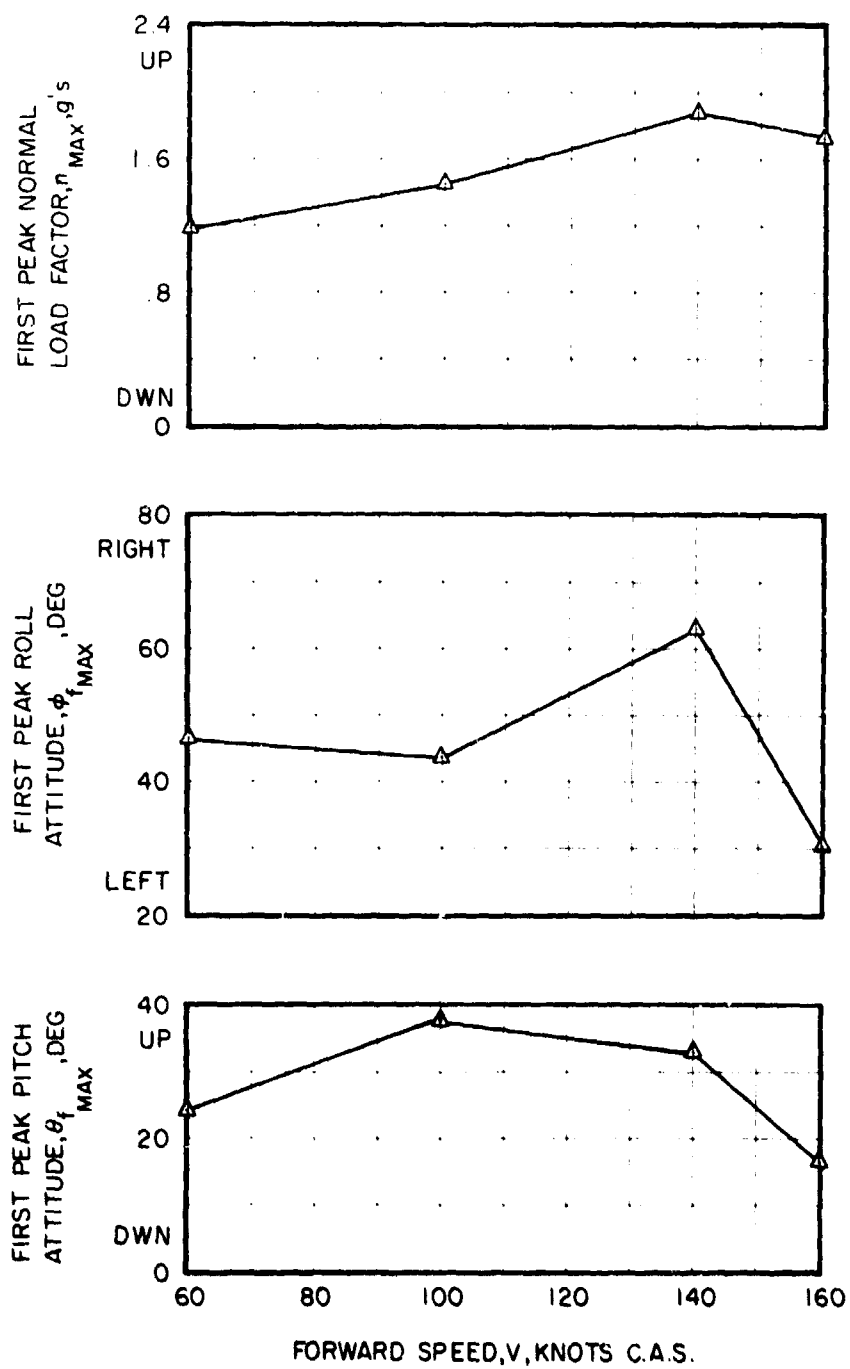
Figure 21.   Concluded.
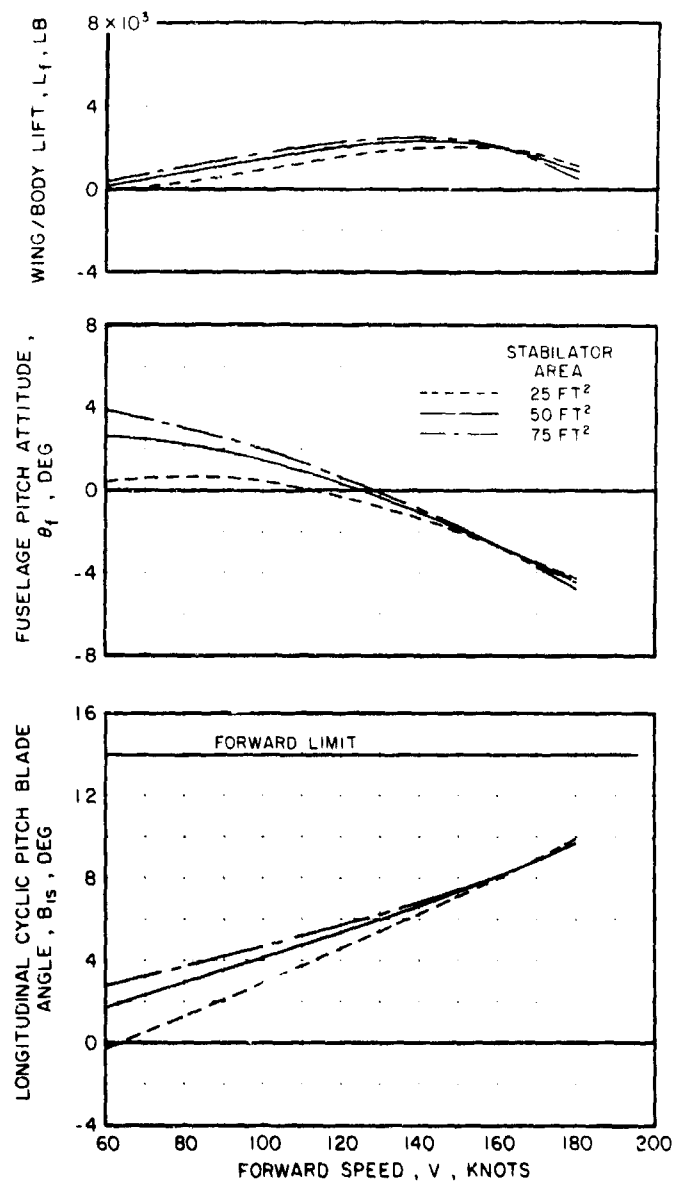
(a) FORWARD C.G. (258 IN.)

Figure 22. Effect of Stabilator Area on Calculated Longitudinal
Trim Characteristics in High-Speed Level Flight for
Forward and Aft C.G. Positions; Base Area = 50 Ft²,
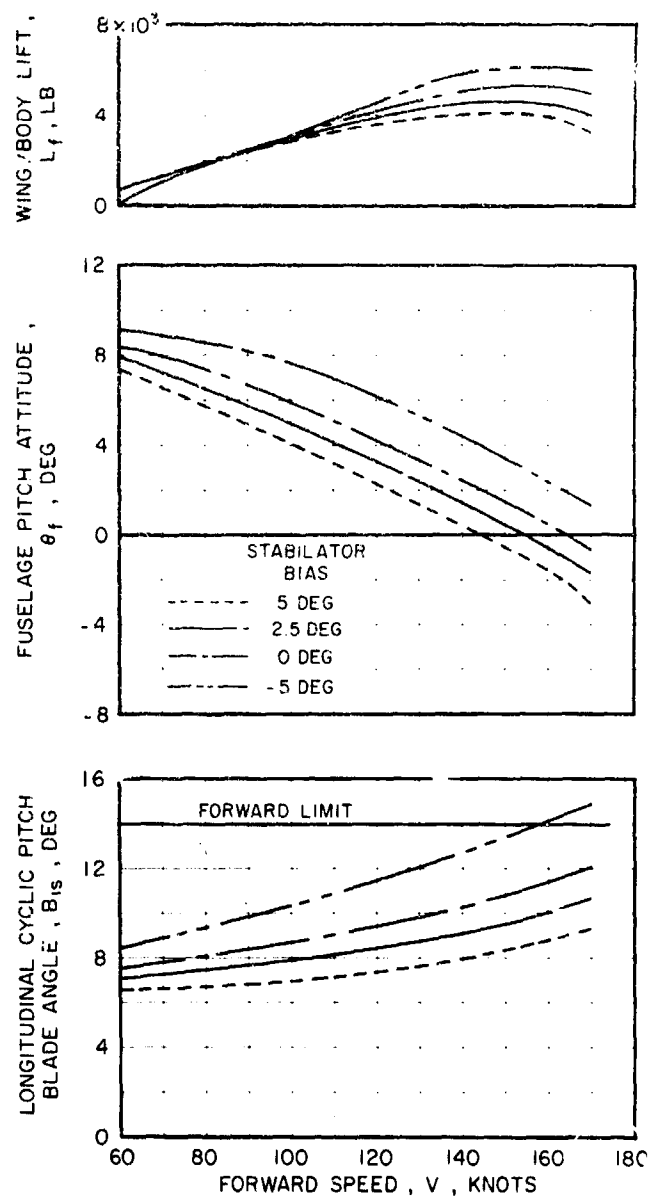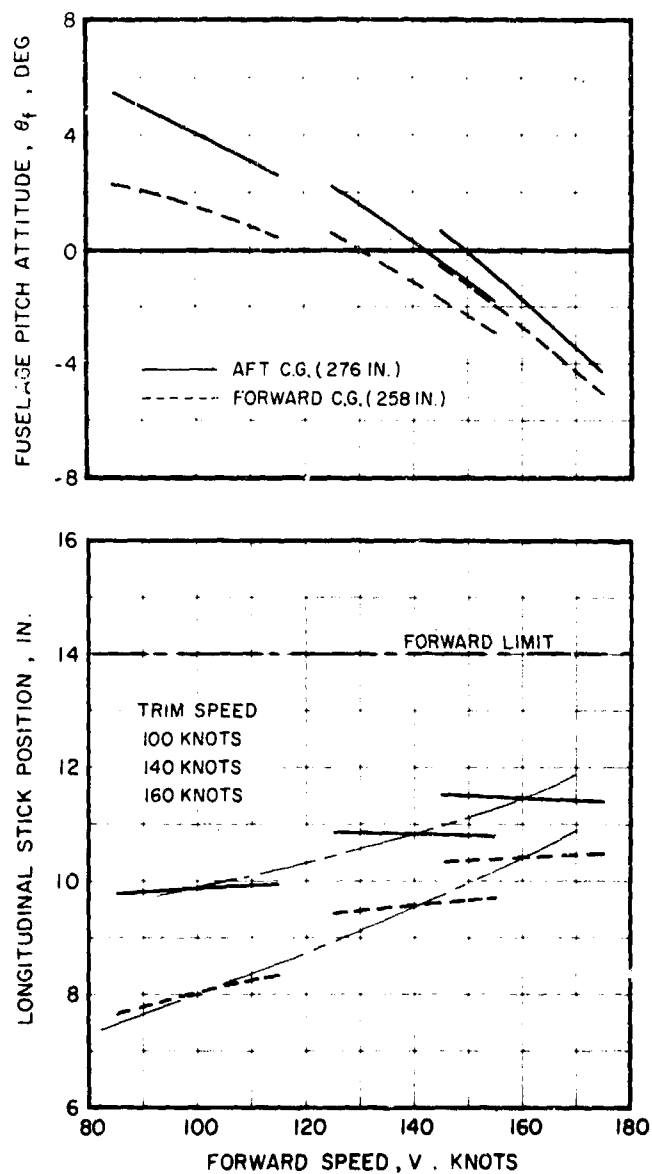Aspect Ratio = 4.8, Bias = 2.5 Deg, Coupling Ratio = 1.0,
G.W. = 17,300 Lb.
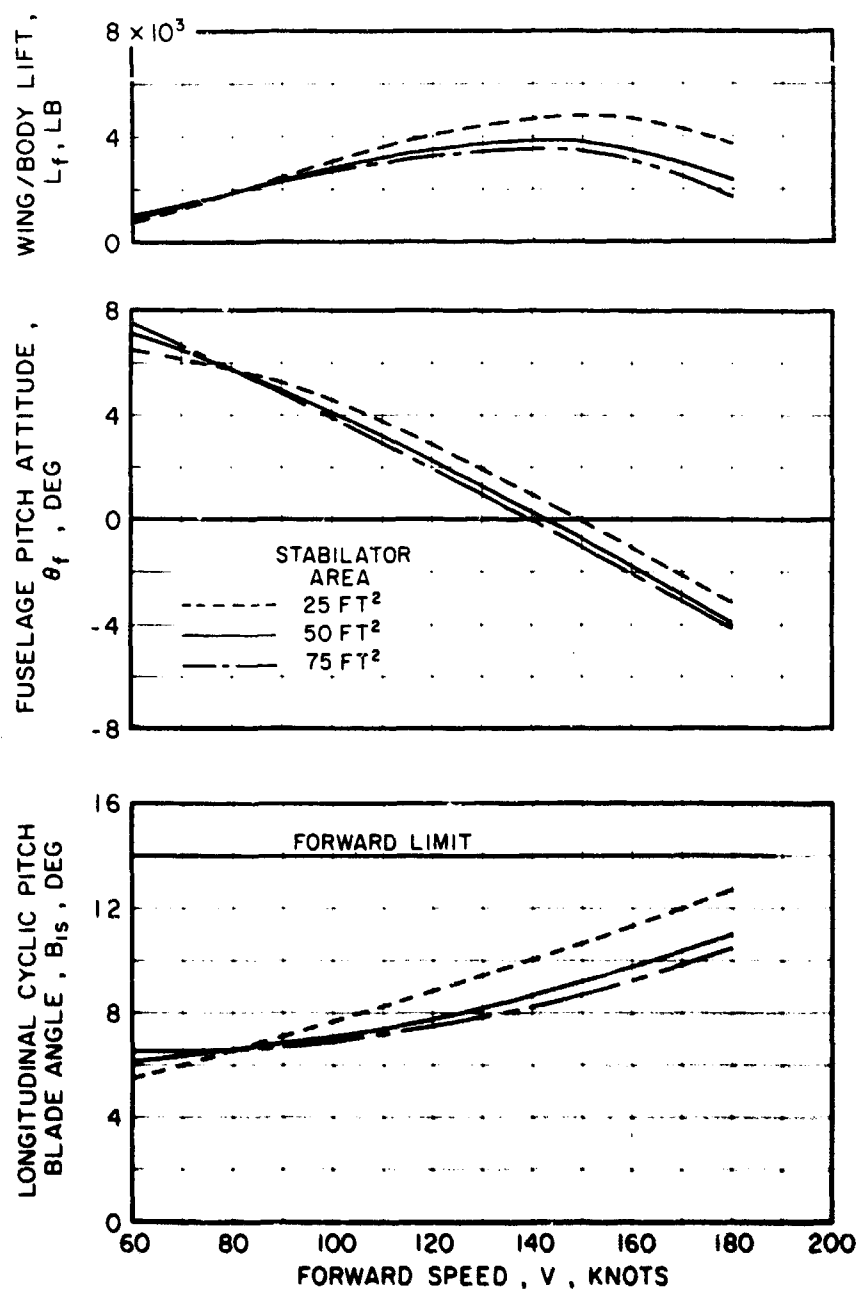
(b) AFT C.G. (276 IN.)

Figure 22. Concluded.

62
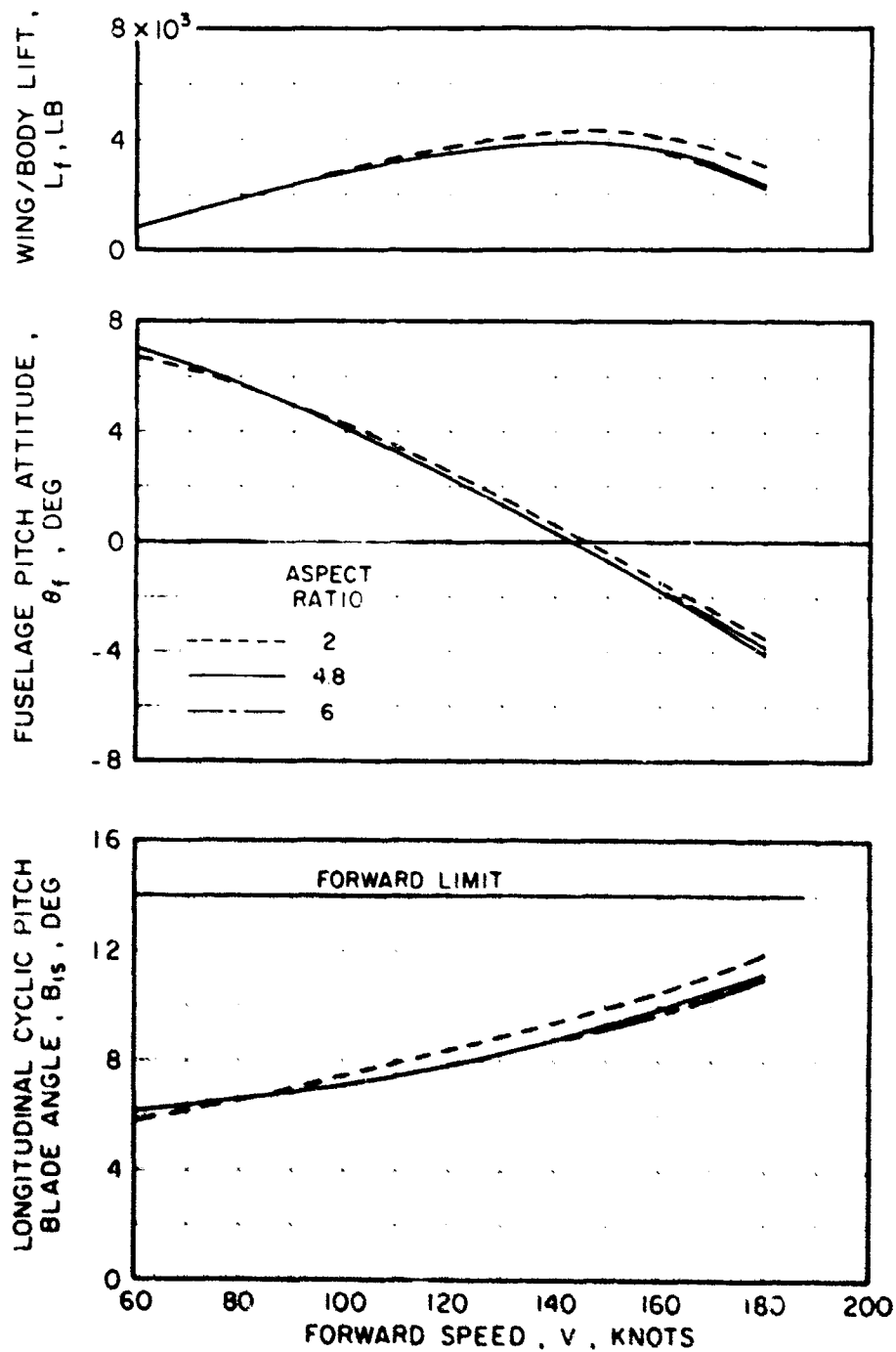
Figure 23. Effect of Stabilator Aspect Ratio on Calculated Longitudinal
Trim Characteristics in High-Speed Level Flight;
Base Aspect Ratio = 4.8, Area = 50 Ft², Bias = 2.5 Deg,
Coupling Ratio = 1.0, G.W. = 17,300 Lb, C.G. = 276 In.
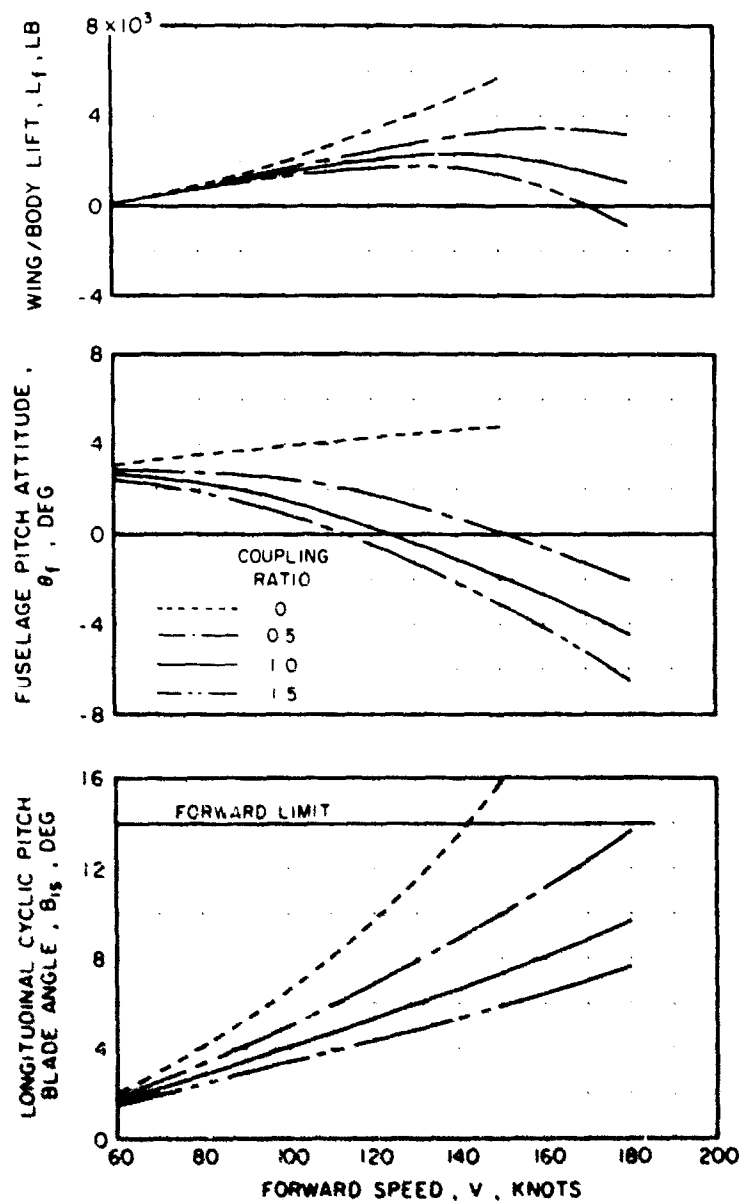
(o) FORWARD C G (258 IN)

Figure 34. Effect of Stabilator Control Coupling Ratio on
Calculated Longitudinal Trim Characteristics in
High-Speed Level Flight for Forward and Aft C.G.
Positions; Base Coupling Ratio = 1.0, Area = 50 Ft²,
Aspect Ratio = 4.6, Bias = 2.5 Deg,
G.W. = 17,300 Lb.

64

(b) AFT C.G (276 IN)

Figure 4. Concluded.
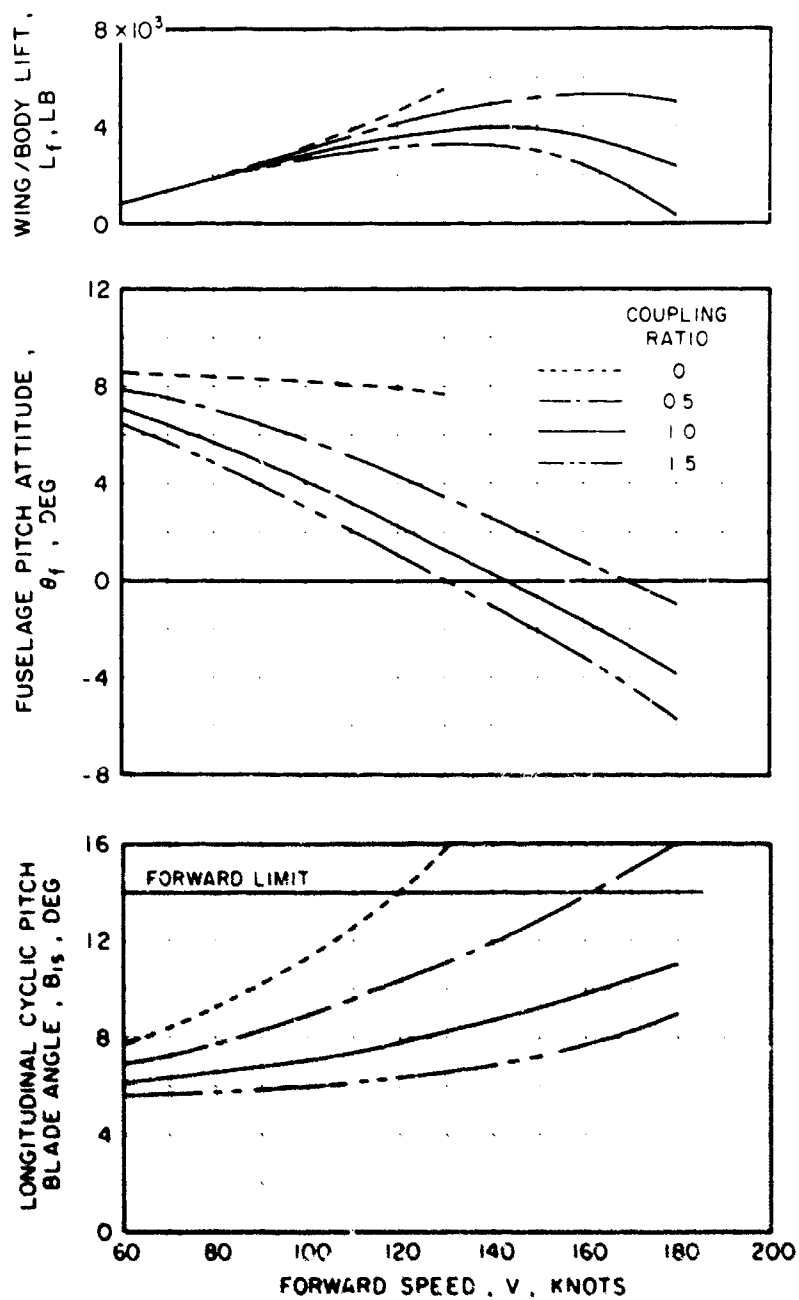
(a) FORWARD C.G. (258 IN.)

Figure 25. Effect of Gross Weight on Calculated Longitudinal
Trim Characteristics in High-Speed Level Flight
for Forward and Aft C.G. Positions; Area = 50 Ft^2,
Aspect Ratio = 4.8, Bias = 2.5 Deg, Coupling Ratio = 1.0.

66

(b) AFT C.G. (276 IN.)

Figure 25. Concluded.

67

(a) FORWARD C.G. (258 IN.)

Figure 26. Effect of Stabilator Bias Setting on Longitudinal
Trim Characteristics in High-Speed Forward Flight
for Forward and Aft C.G. Positions; Base Bias
Setting = 2.5 Deg, Area = 50 Ft^2, Aspect Ratio = 4.8,
Coupling Ratio = 1.0, G.W. = 22,000 Lb.

(b) AFT C.G. (276 IN.)

Figure 26. Concluded.

(a)

Figure 27. Calculated Longitudinal Static Stability Character-
istics of the S-67 Aircraft with Base Stabilator
Design in High-Speed Flight for Forward and Aft
C.G. Positions; Area = 50 Ft^2, Aspect Ratio = 4.8,
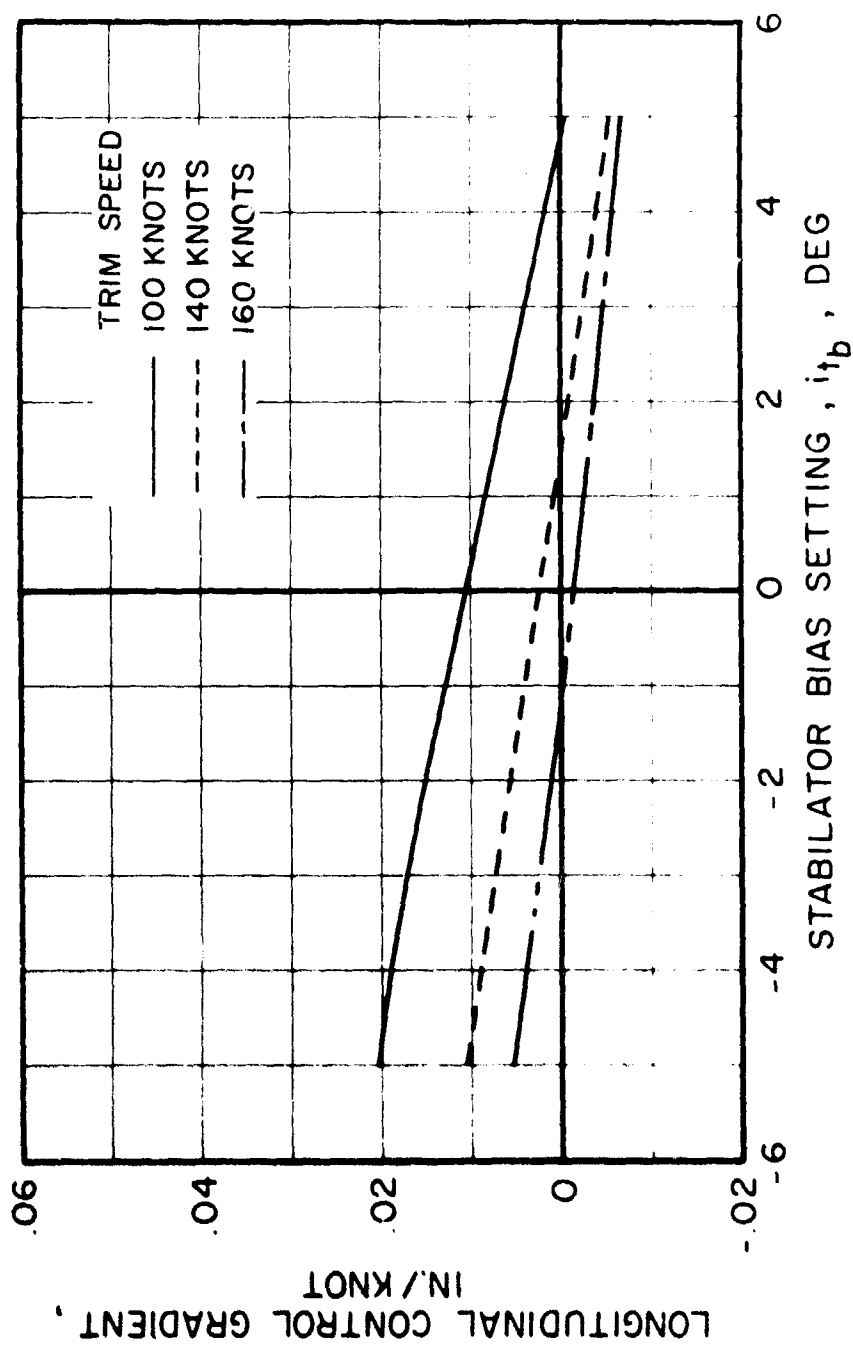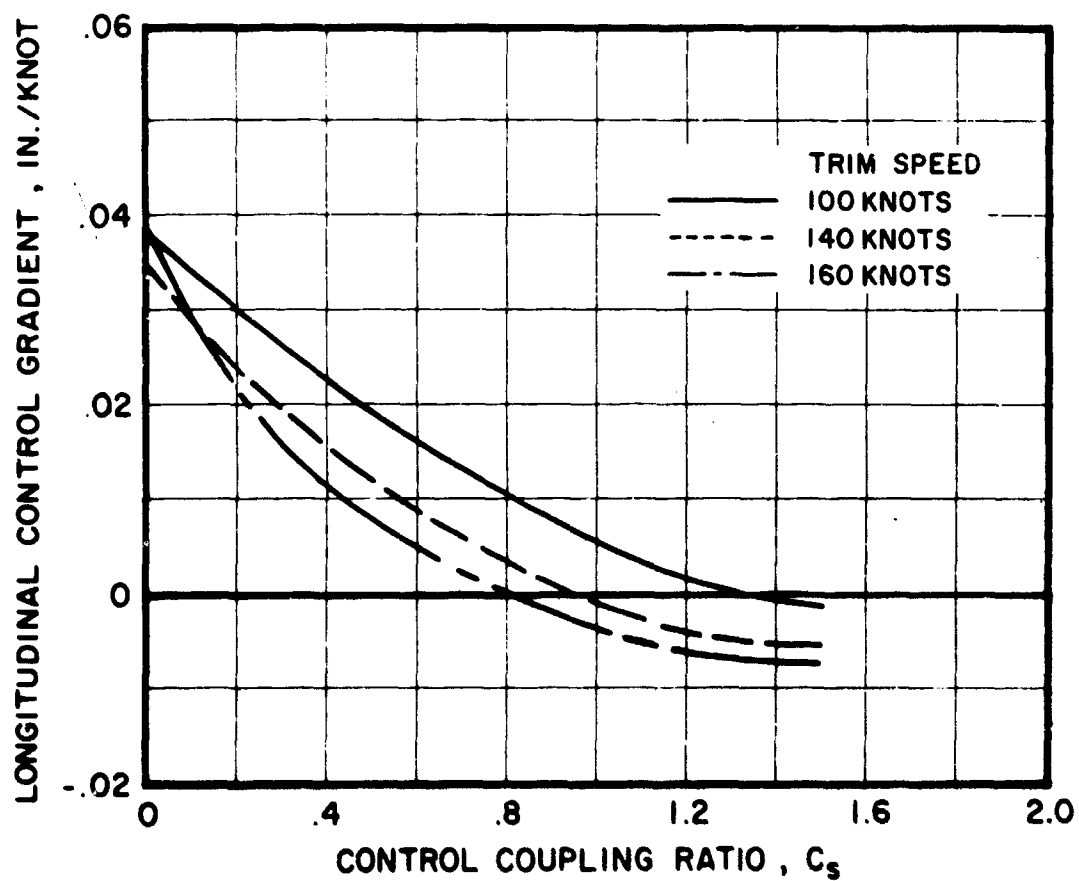Bias = 2.5 Deg, Coupling Ratio = 1.0, G.W. = 17,300 Lb.
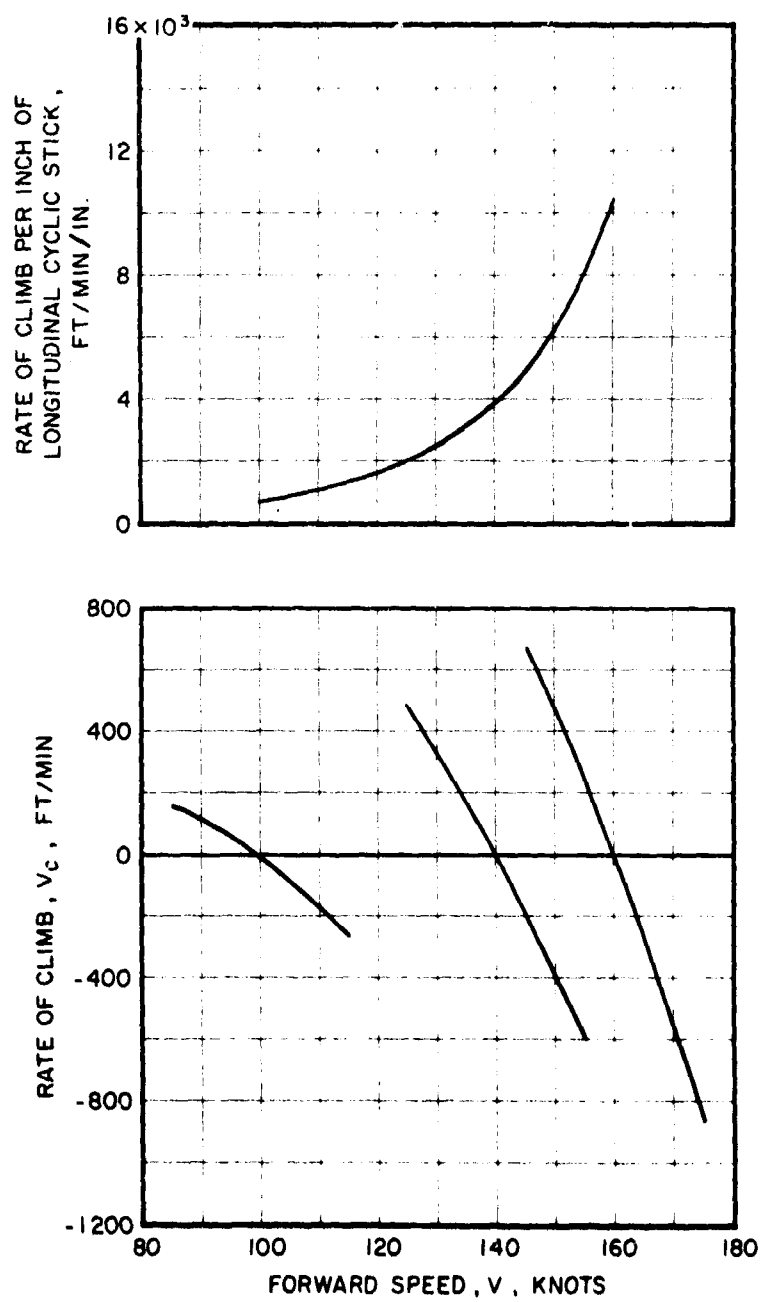
(b)

Figure 27. Concluded.

71
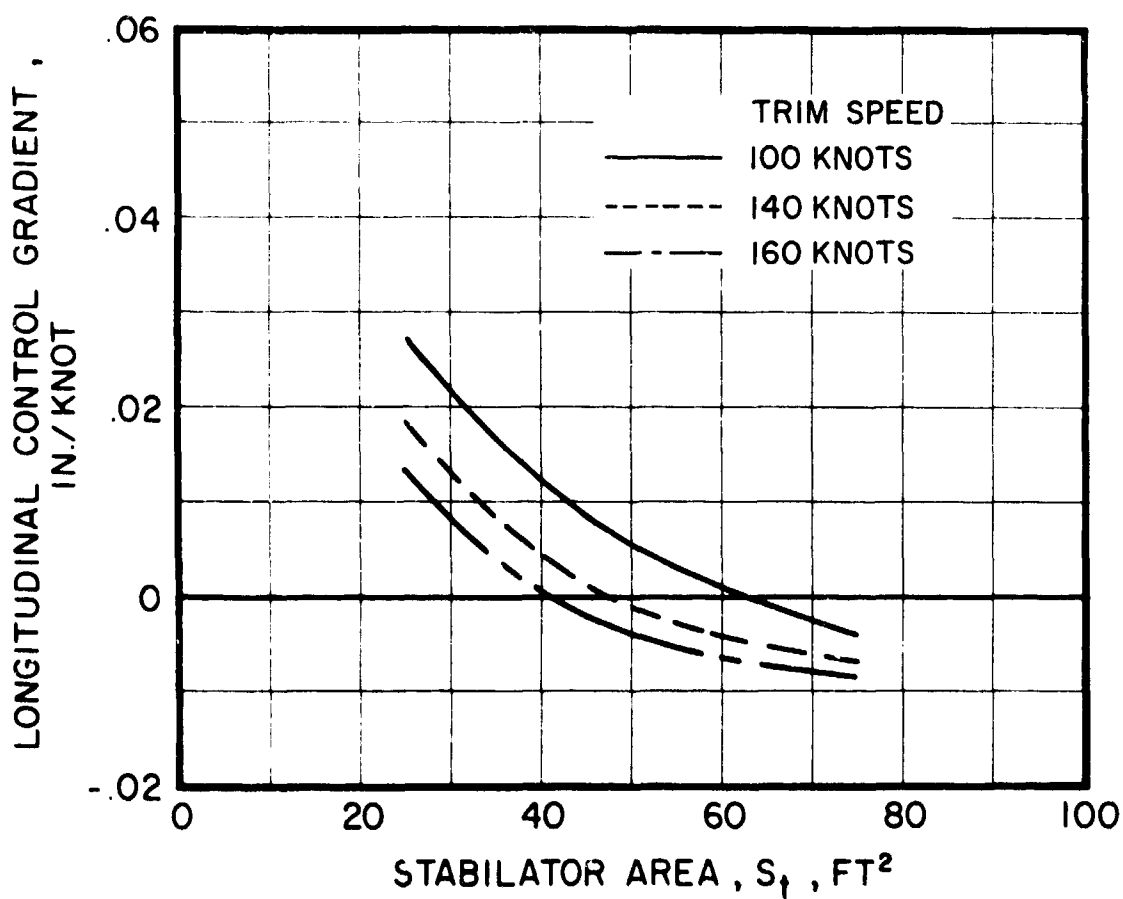
Figure 28. Effect of Stabilator Area on Calculated Static
Stability Longitudinal Control Gradient in High-
Speed Flight; Base Area = 50 Ft^2, Aspect Ratio =
4.8, Bias = 2.5 Deg, Coupling Ratio = 1.0,
G.W. = 17,300 Lb, C.G. = 276 In.

Figure 9. Effect of Stabilator Bias Setting on Calculated Static Stability Longitudinal Control Gradient in High-Speed Flight; Base Bias Setting = 2.5 Deg, Area = 50 Ft², Aspect Ratio = 4.8, Coupling Ratio = 1.0, G.W. = 17,300 Lb, C.G. = 276 In.

Figure 30. Effect of Stabilator Control Coupling Ratio on
Calculated Static Stability Longitudinal Control
Gradient in High-Speed Flight; Base Coupling
Ratio = 1.0, Area 50 Ft^2, Aspect Ratio = 4.8,
Bias = 2.5 Deg, G.W. = 17,300 Lb, C.G. = 276 In.

74
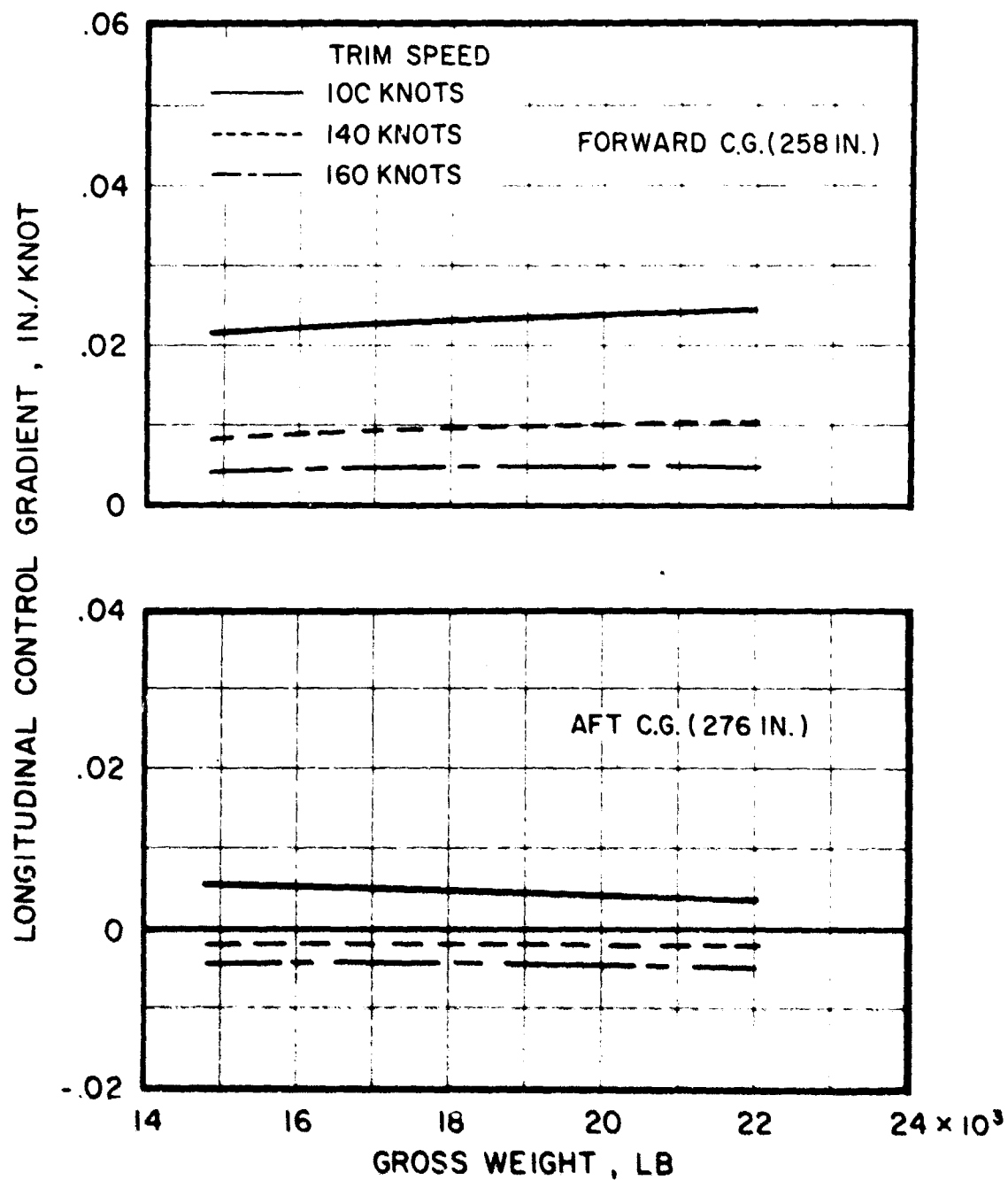
Figure 31. Effect of Gross Weight on Calculated Static Stability
Longitudinal Control Gradient in High-Speed Flight
for Forward and Aft C.G. Positions; Area = 50 F+²,
Aspect Ratio = 4.8, Bias = 2.5 Deg, Coupling Ratio = 1.0.

75
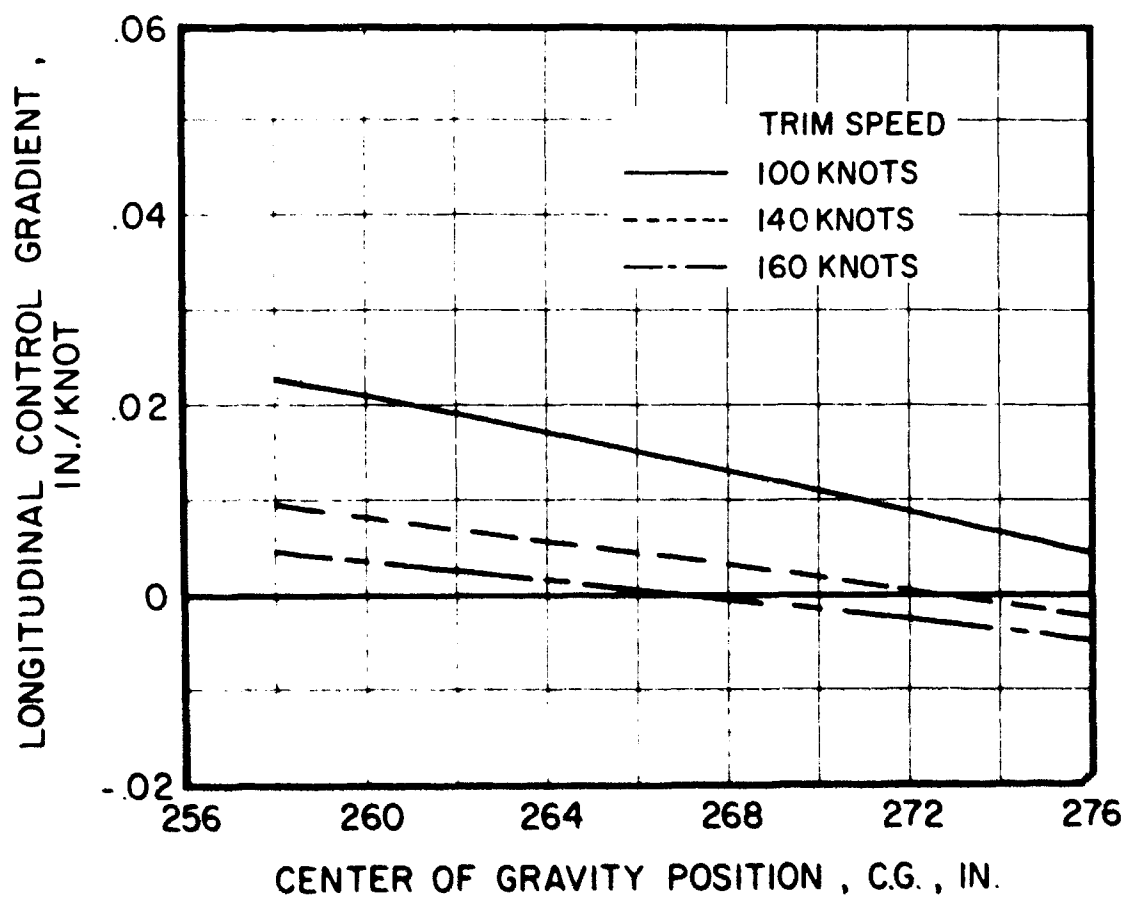
Figure 36. Effect of Center-of-Gravity Position on Calculated Static Stability Gradient in High-Speed Flight; Area = 50 Ft², Aspect Ratio = 4.8, Bias = 2.5 Deg, Coupling Ratio = 1.0, G.W. = 17,300 Lb.
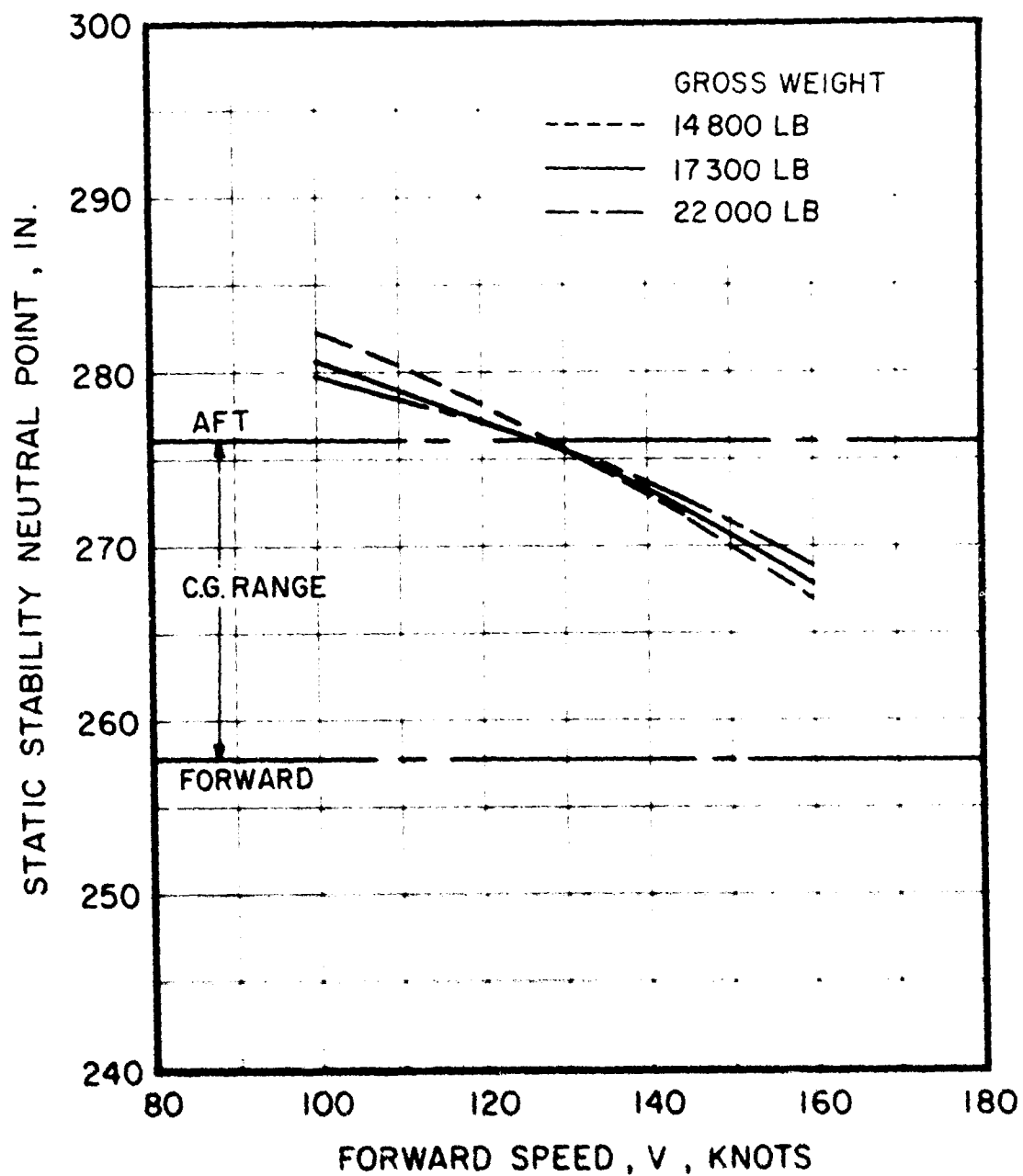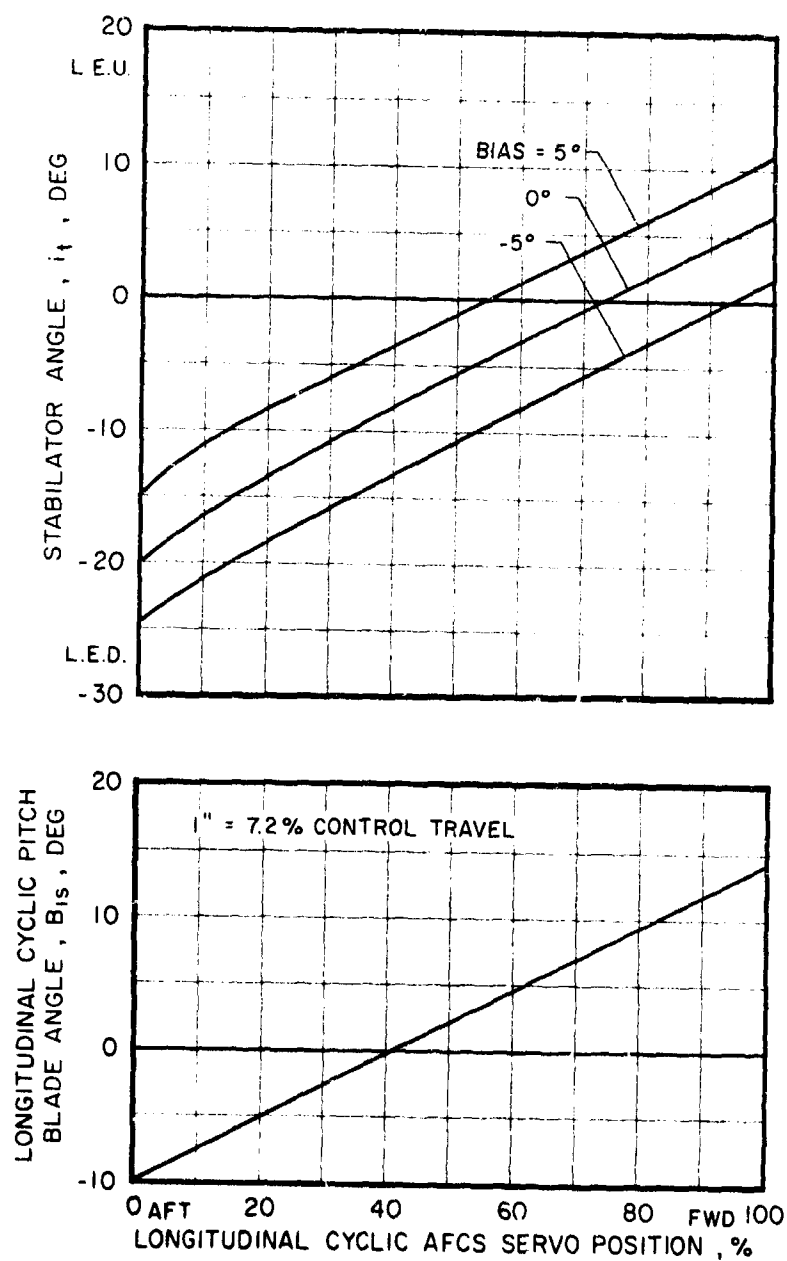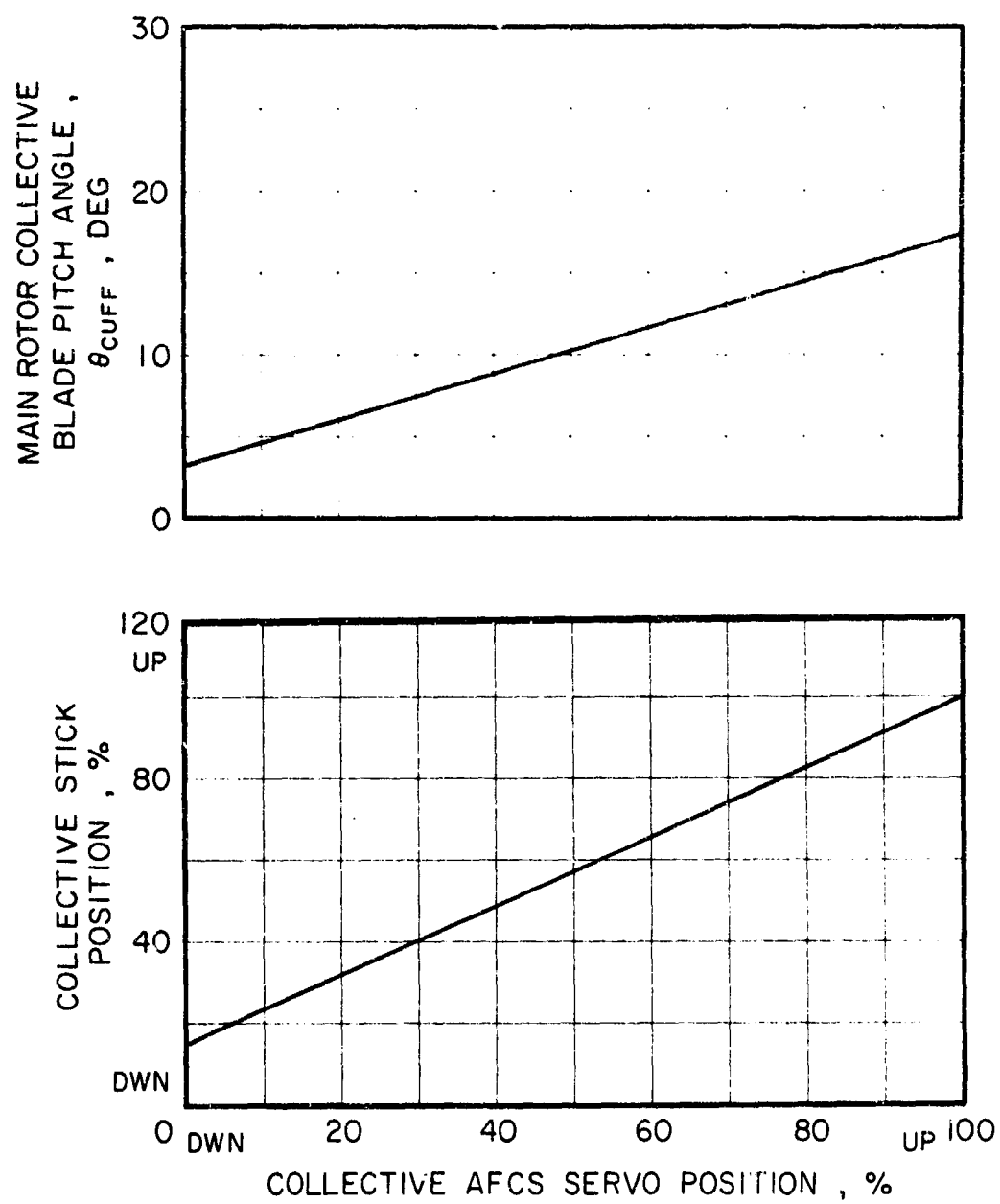
Figure 37 Variation of Calculated Static Stability Neutral
Point With Forward Speed for Several Aircraft
Gross Weights; Stabilator Area = 50 Ft², Aspect
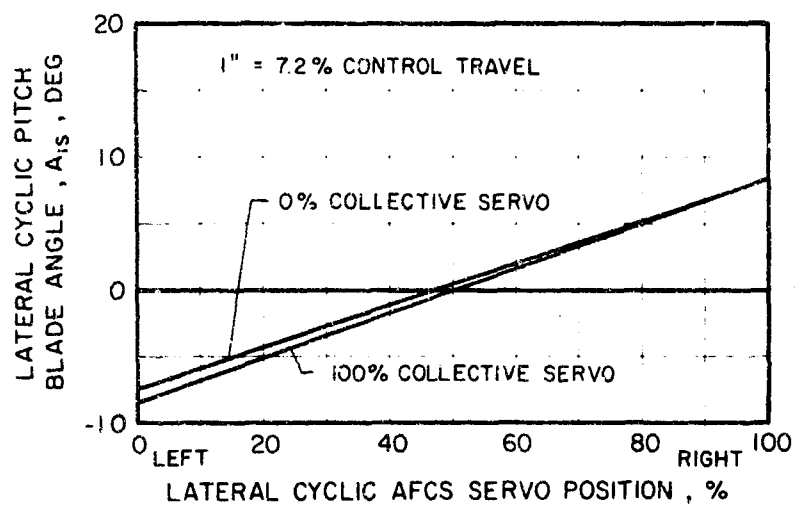Ratio = 4.8, Bias = 0.5 Deg, Coupling Ratio = 1.0.

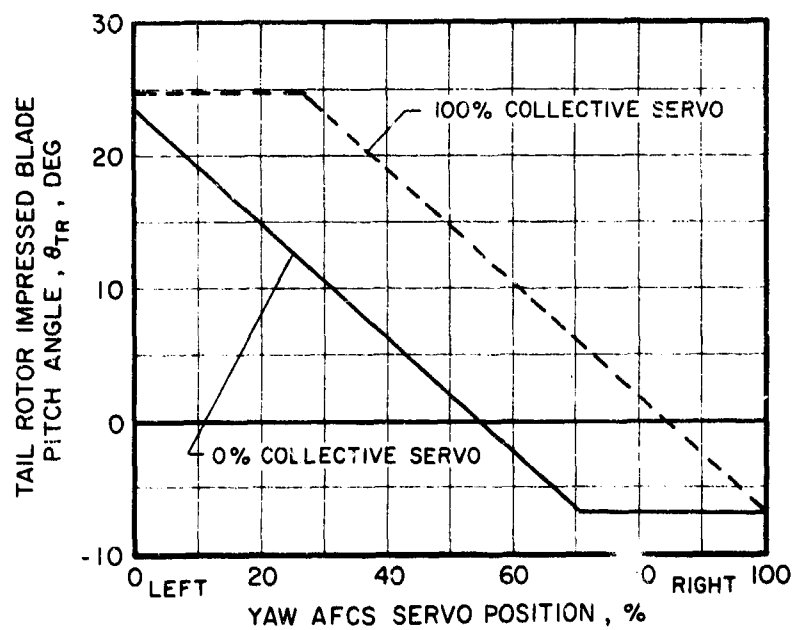(a) LONGITUDINAL CONTROL RIGGING

Figure 34. Flight Control Rigging.

80

(b) COLLECTIVE CONTROL RIGGING

Figure 34. Continued.

81

(c) LATERAL CONTROL RIGGING

(d) YAW CONTROL RIGGING

Figure 34. Concluded.

82

SIMULATION DESCRIPTION AND CORRELATION

A computer simulation program for the S-67 aircraft was developed to evaluate design changes in the stabilator system. The General Helicopter Simulation Program [5] was programmed on the Digital Equipment Corporation PDP-10 digital computer for the simulation study. Aircraft dimensions and aerodynamic data from wind tunnel tests were used to develop the simulation of the S-67 aircraft. The computer program was then modified to improve the accuracy of the main rotor hub calculation and to simulate the phase angle between swashplate motion and rotor blade pitch angle motion actually on the aircraft. Also, synthesized aerodynamic pitching moment data were developed to predict the effects of stabilator design changes. Prior to its use in the simulation study, the simulation program was up-dated and confirmed with flight test results.

Simulation Program Description

The S-67 aircraft was simulated by aerodynamic, inertial, and mass forces and moments in the conventional translation and rotation equations of motion in six degrees of freedom. The rotor had two additional degrees of freedom: flapping and rotor speed. Wind tunnel data from a 1/12 scale model test [6] of the S-67 was used to describe the aerodynamic force and moment contribution of the fuselage and lifting surfaces. The longitudinal wind tunnel data (lift, drag, and pitching moment) were functions of fuselage angle of attack for various stabilator incidence angles. The lateral data (side force, rolling moment, and yawing moment) were defined in terms of sideslip angle. The test data were analytically extrapolated to ±30 degrees for the primary independent variables (angle of attack and sideslip). The ±12 degrees test range for stabilator angle, however, was retained.

The aerodynamic forces and moments of the rotor were calculated by the standard blade element analysis at discrete blade stations. Two-dimensional flow was assumed at each blade station. The local blade forces and moments were integrated along the blade span and the loads on each blade were summed to calculate the total rotor forces and moments. The simulated rotor had 5 rigid blades with each blade divided into 4 segments for the blade element analysis. The local lift and drag coefficients at each blade segment were obtained from wind-tunnel-derived airfoil section aerodynamic data. The nonlinear, steady-state data included stall and compressibility effects so that no limitations were imposed on the magnitude of advance ratio, inflow angle, rotor blade local angle of attack, or Mach number.

The trim and static stability of the aircraft were obtained by a repetitive calculation of the equations of motion. The calculation procedure was initiated with an assumed pitch attitude and the trim airspeed. The forces and moments on the rotor and body were then calculated, and the equations of motion were integrated to obtain the aircraft accele..., velocities and attitudes. The control parameters were adjusted and the calculation cycle was repeated until all accelerations were zero and the simulated aircraft was trimmed.

To permit variations in stabilator design, such as area and aspect ratio, aircraft pitching moment data were synthesized to include an analytical representation of the stabilator. DATCOM [7] procedures were used to predict the stabilator contribution. Trim characteristics calculated by the simulation program using, first, the actual wind tunnel data and, then, the synthesized data were compared to verify the accuracy of the analytical technique (Figure 35). The good agreement between the calculated data indicate that the synthesized data can be used to predict the effects of stabilator design parameter changes.

Correlation With Flight Test Data

The computer program was further substantiated by correlating calculated trim performance data with flight test results and updated by introducing empirical corrections to improve the accuracy of the calculated data. The results of the correlation study are presented in Figures 36 through 40. Data for the light-gross-weight (14,800 lb), aft c.g. (276 in.) condition are presented in Figures 36, 37, and 38 for stabilator bias angle of 0, 2.5, and 5.0 degrees respectively. Data for the heavy-gross-weight (17,300 lb) loading with a bias angle of 0 degrees are presented in Figures 39 and 40 for forward c.g. (258 in.) and aft c.g. (276 in.) respectively. In addition to the high-speed flight data, hover data for the light and heavy gross weights are also included in Figures 36, 39, and 40. In these figures, the calculated values at 80 knots and hover are connected by a dashed line to indicate that no correlation was attempted in this low-speed region.

The results in Figures 36 and 40 indicate good correlation between the calculated values and flight test results for aircraft attitude and flight control parameters. Above 180 knots, however, the simulation program predicts an early power rise due to rotor stall effects. Modification to the program, such as rotor blade spanwise flow effects and nonsteady airfoil section aerodynamic data, are currently being investigated to improve rotor performance prediction methods in high-speed flight.

Several minor empirical modifications, however, were made to improve the accuracy of the simulation program. These include:

1. A main rotor blade twist correction to account for the aeroelastic torsional twisting of the swept blade-tip.

2. A 2-inch lateral displacement of the aircraft c.g. position to the left to account for asymmetrical loading at the light-gross-weight condition.

3. A 2-degree leading-edge-up stabilator bias correction to account for main rotor downwash effects.

4. A nose-left correction ($N/q = -150$ ft^3) to the wind tunnel yawing moment data to account for the variation in yawing moment with pitch attitude and the interference effects between the main rotor, tail rotor and vertical stabilizer.

The main rotor twist correction was programmed as a function of forward speed. The existing -4 degree linear blade twist was increased by -2 degrees (to -6 degrees) for hover and 80 knots forward speed. From 80 knots, the twist correction was reduced linearly with forward speed to a value of -0.5 (total twist = -4.5 degrees) at 180 knots. Although this empirical twist correction is a first-order approximation for aeroelastic blade twist, good correlation between calculated and test values of main rotor collective blade pitch angle was obtained. The lateral c.g. displacement correction was incorporated for the light-gross-weight condition to improve the predicted value of lateral cyclic pitch blade angle. The ballast distribution for the heavy gross weight loading positioned the center of gravity closer to the aircraft plane of symmetry, so a lateral c.g. correction was not required for this condition.

Although the particular values of these empirical modifications can be refined by further iterations, the correlation results, shown in Figures 36 through 40, are acceptable. The simulation program will predict the effects of stabilator design changes on the trim and static stability of the aircraft.
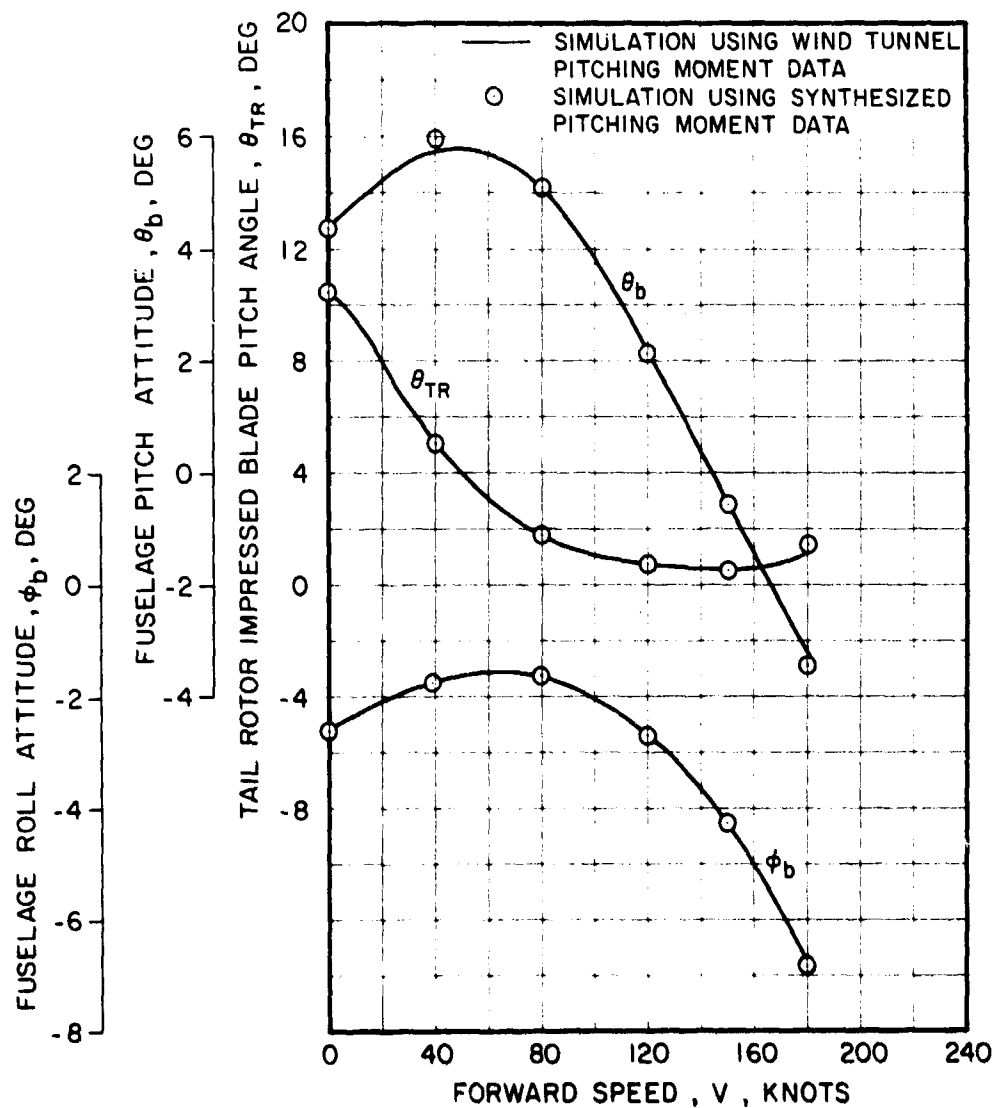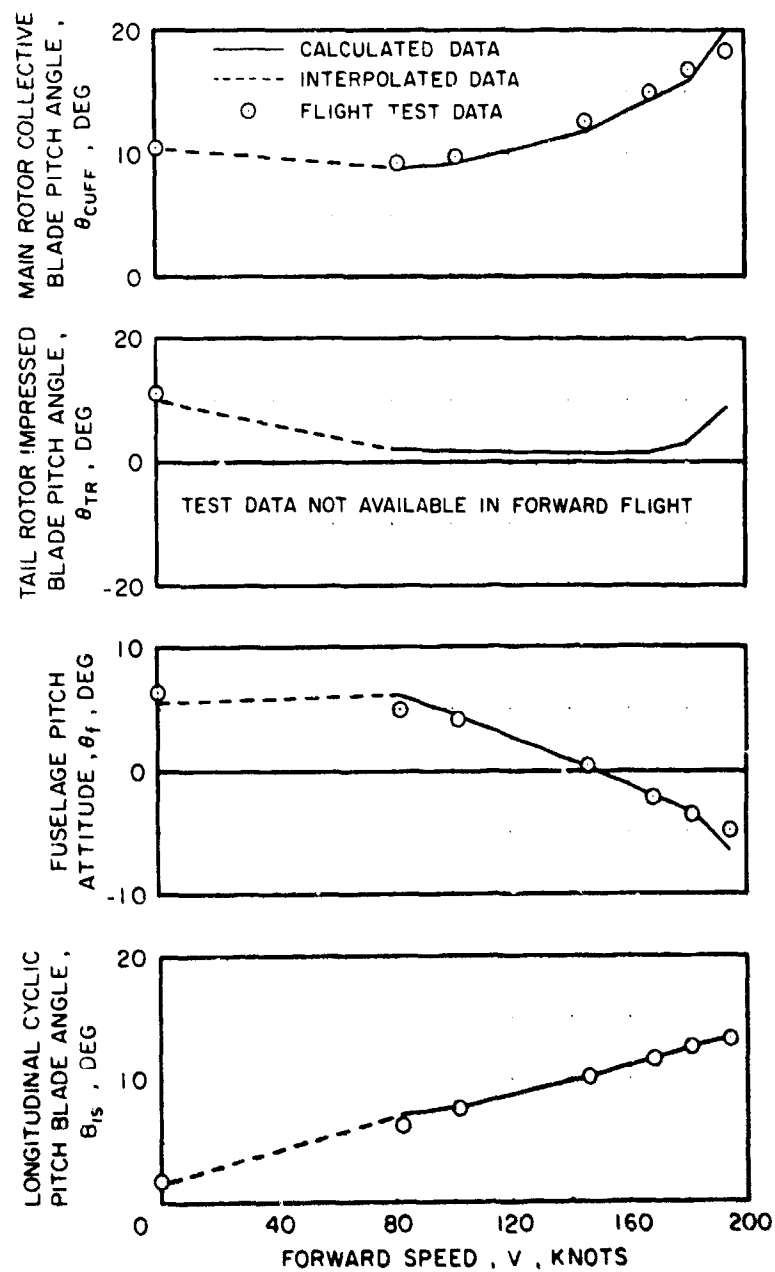
(a)

Figure 35. Comparison of Aircraft Trim Characteristics
            Calculated by Computer Simulation Using Wind
            Tunnel and Synthesized Pitching Moment Data;
            Stabilator Bias = 3.1 Deg, G.W. = 17,200 Lb,
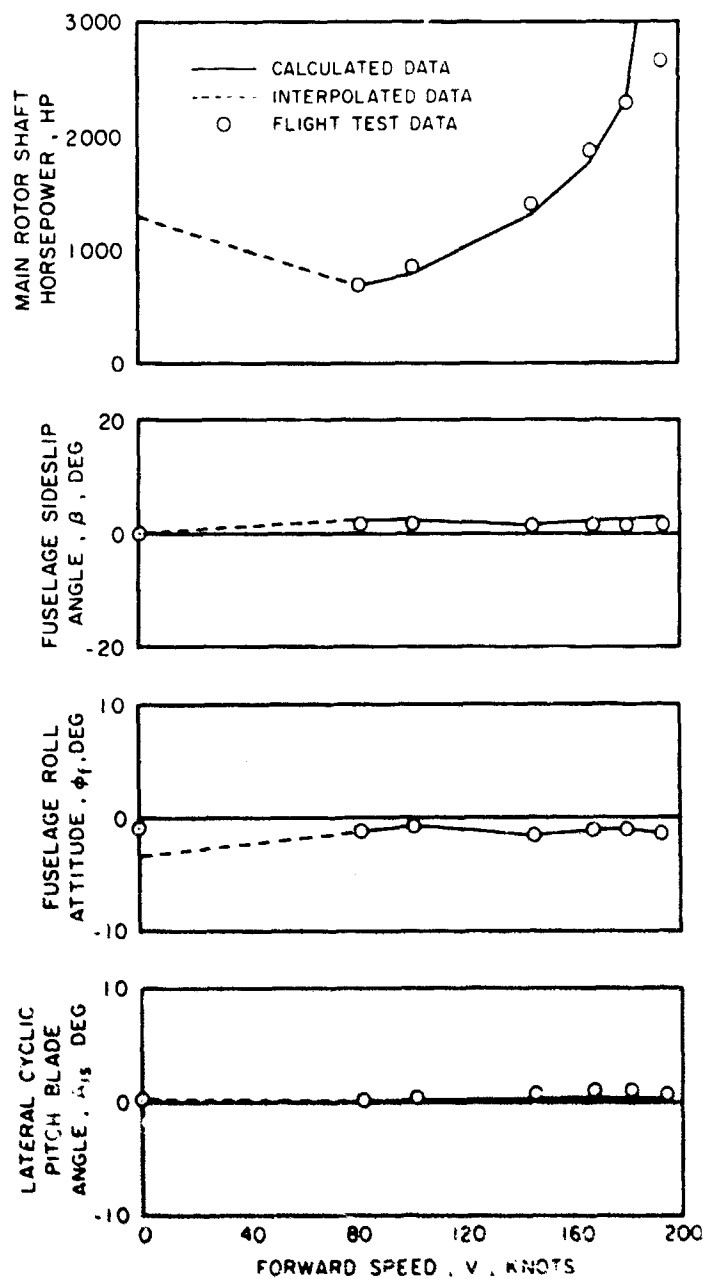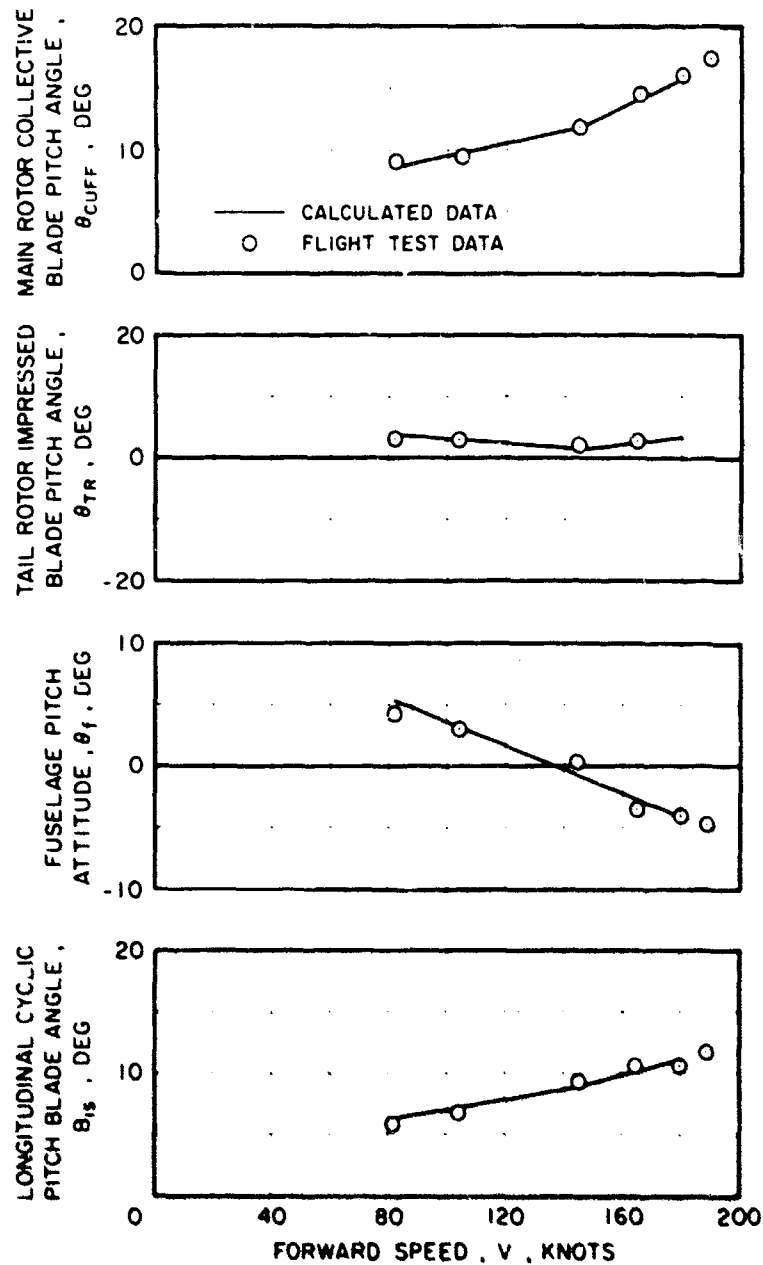   C.G. = 272.5 In.

86
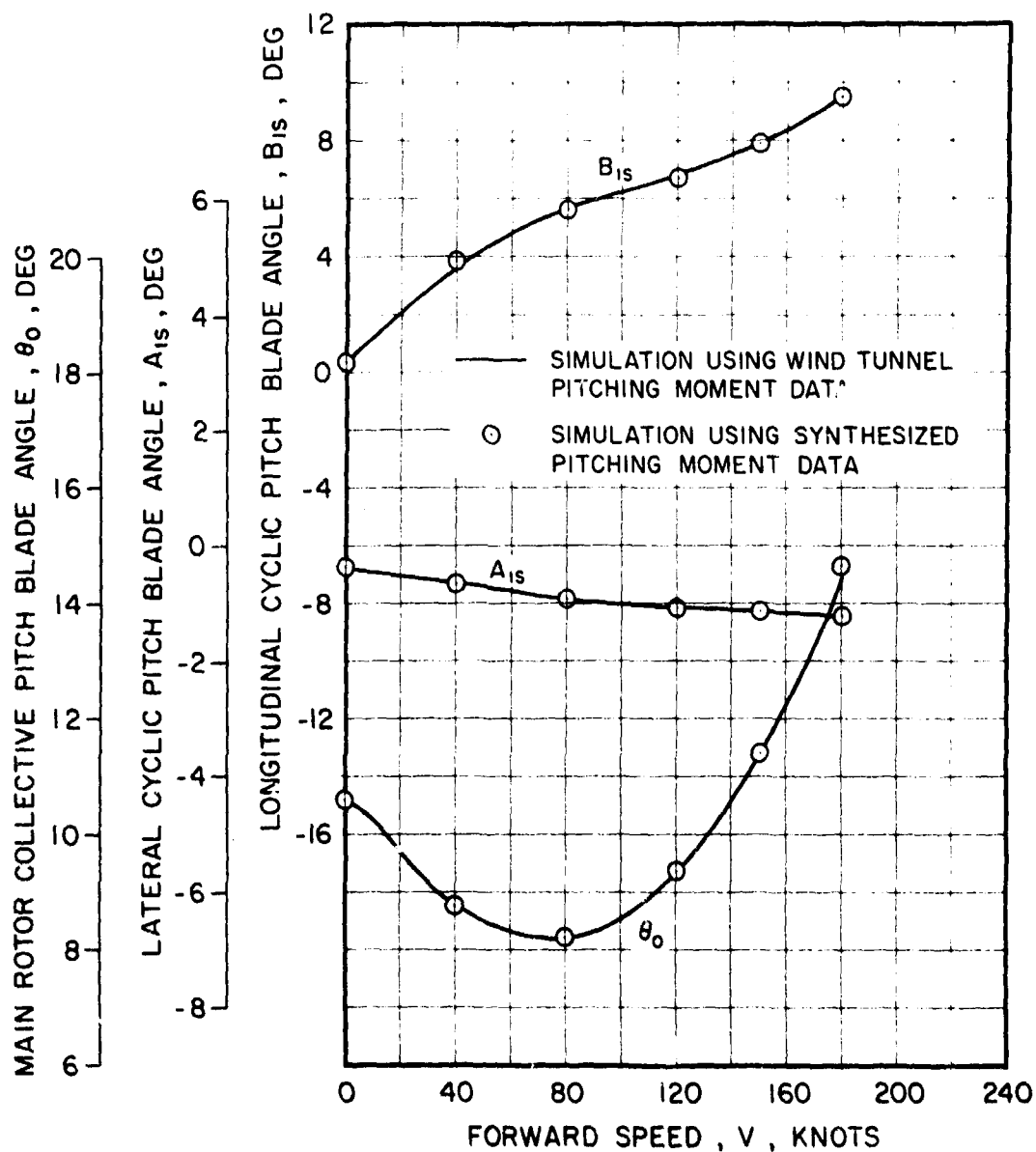
(b)

Figure 35. Concluded.

87

(a) LONGITUDINAL TRIM PARAMETERS

Figure 36. Correlation of Calculated Aircraft Trim
Characteristics With Flight Test Data;
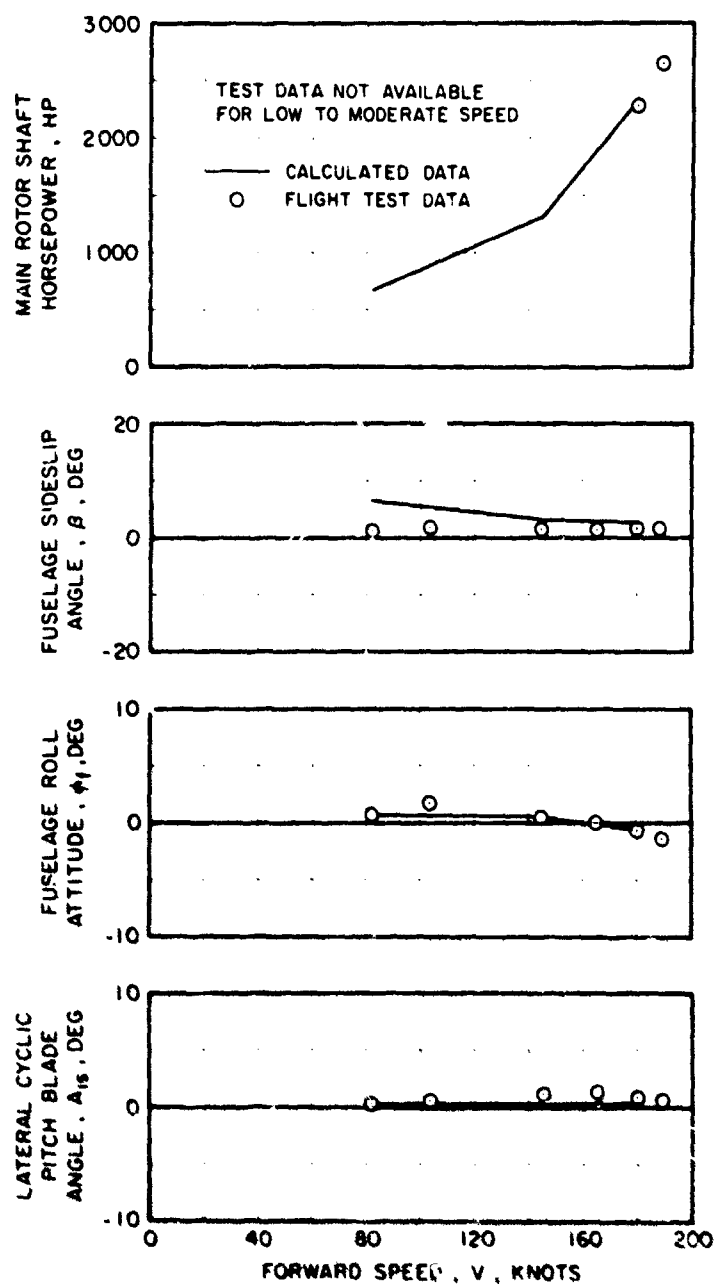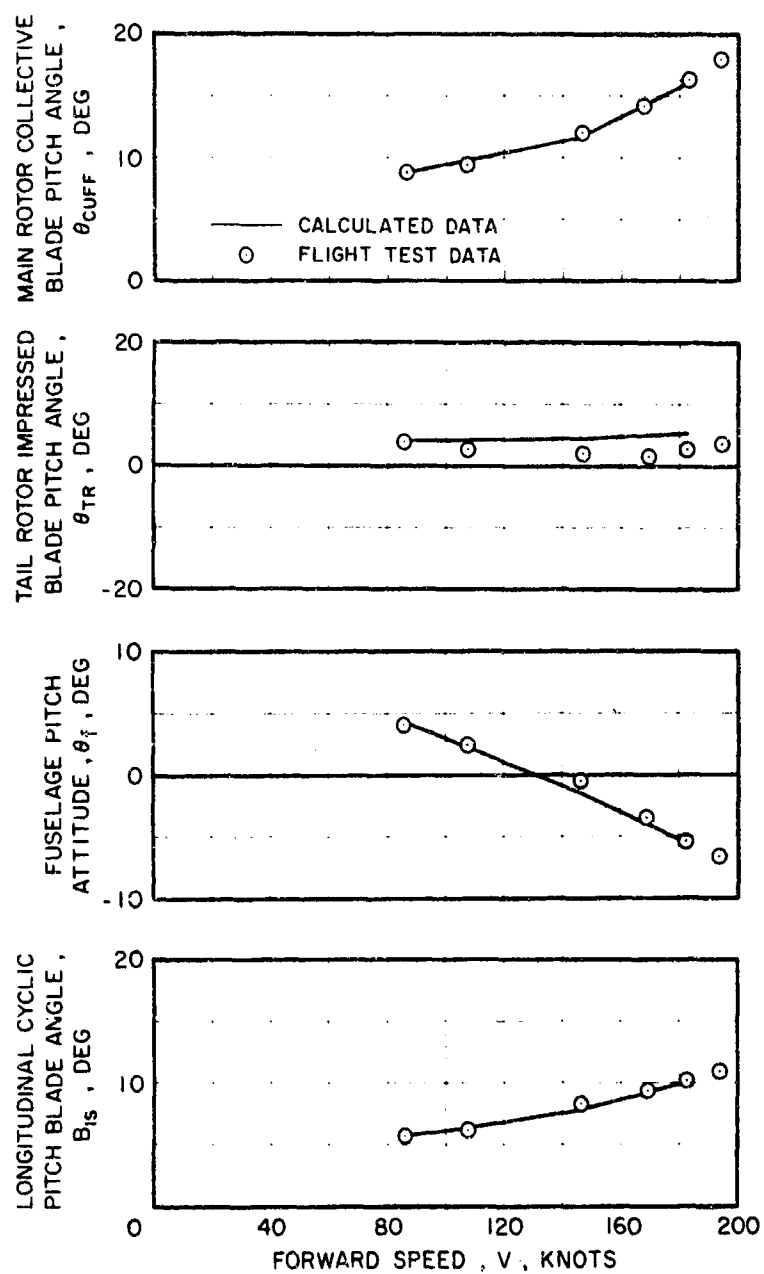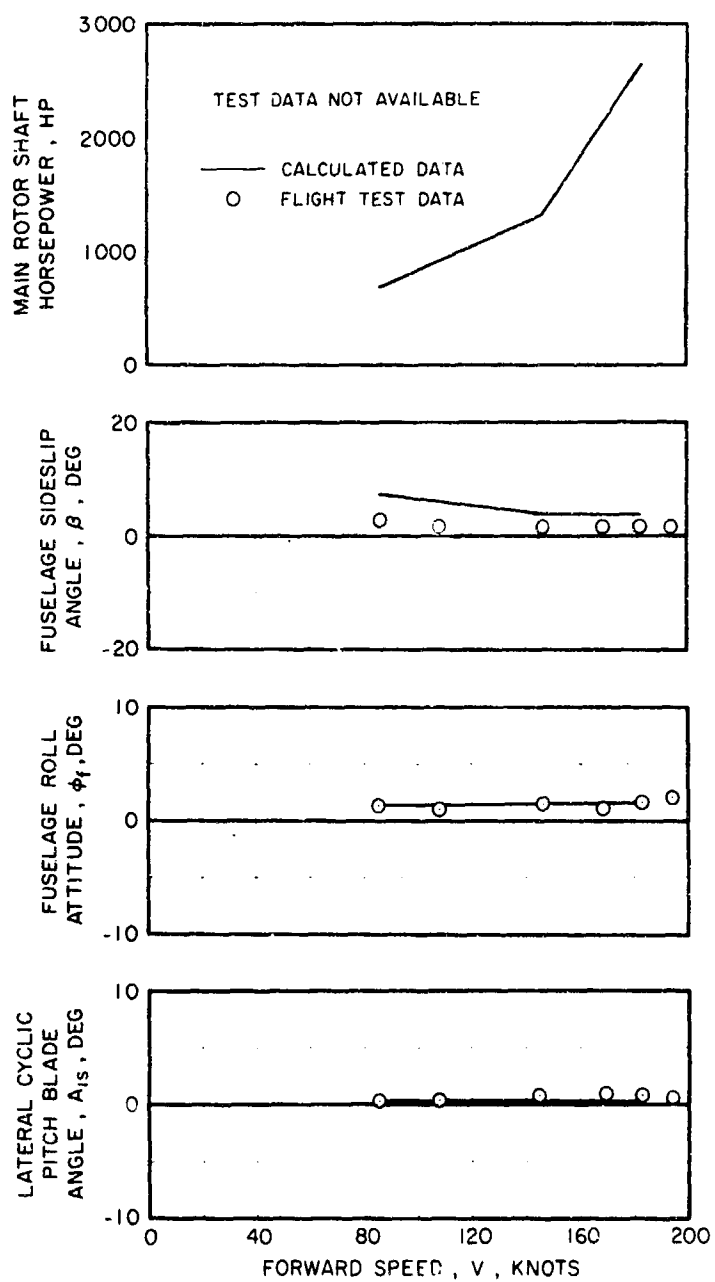Bias = 0 Deg, G.W. = 14,800 Lb, C.G. = 276 In.
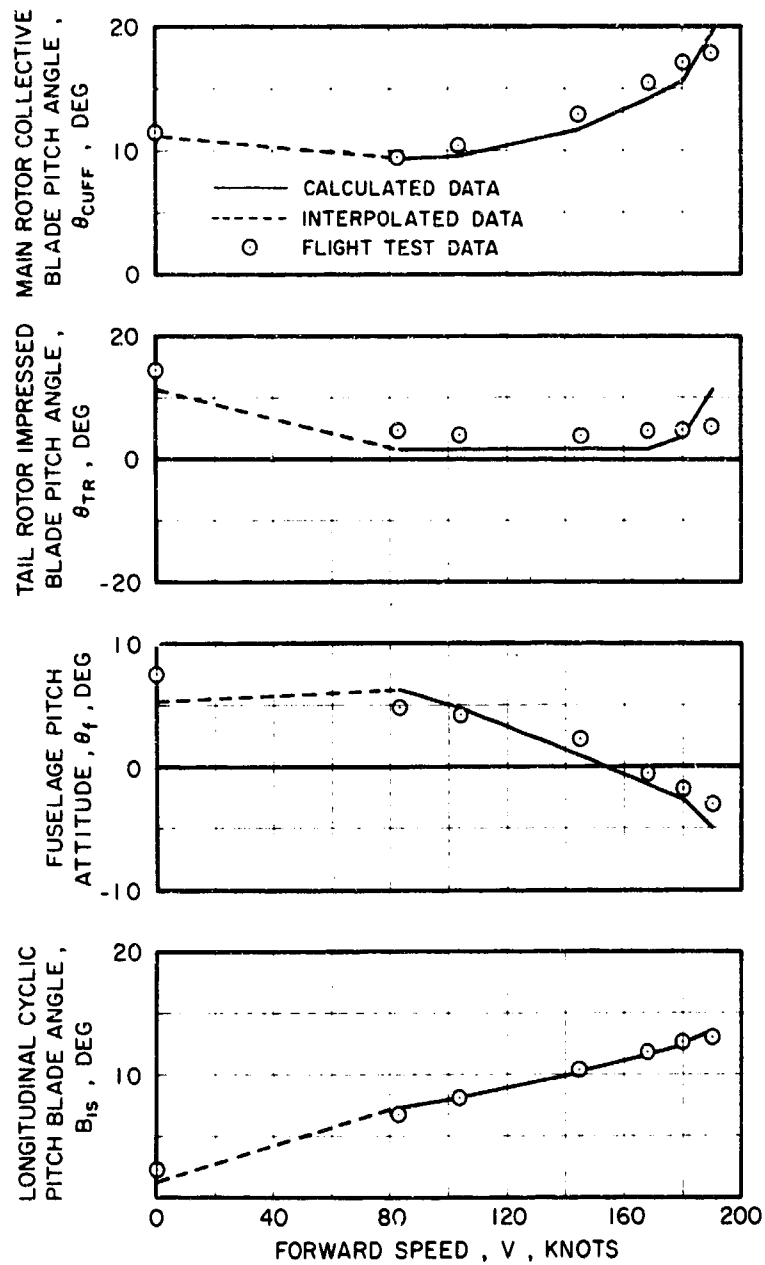
(b) LATERAL TRIM PARAMETERS

Figure 30. Concluded.

(o) LONGITUDINAL TRIM PARAMETERS

Figure 37. Correlation of Calculated Aircraft Trim
Characteristics With Flight Test Data;
Bias = 2.5 Deg, G.W. = 14,400 Lb, C.G. = 276 In.

(b) LATERAL TRIM PARAMETERS

Figure 37. Concluded.

91

(a) LONGITUDINAL TRIM PARAMETERS

Figure 38. Correlation of Calculated Aircraft Trim
Characteristics With Flight Test Data;
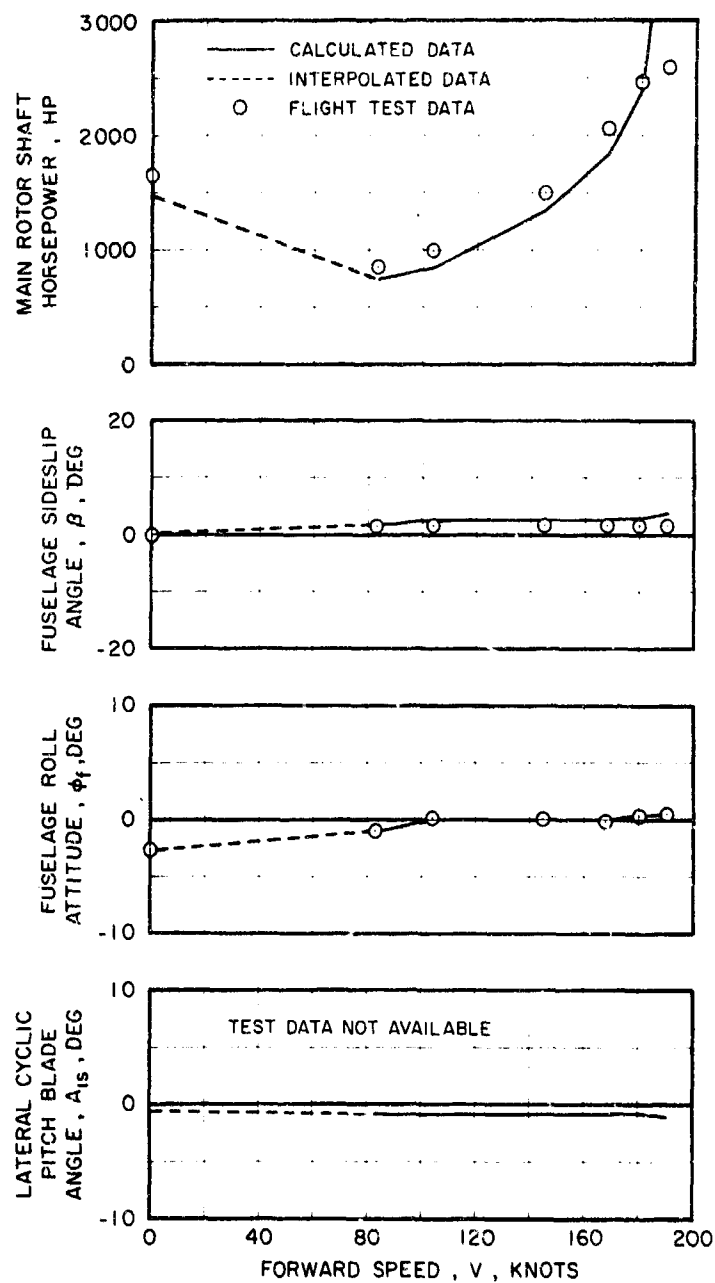Bias = 5.0 Deg, G.W. = 14,800 Lb, C.G. = 276 In.

92

(b) LATERAL TRIM PARAMETERS

Figure 38. Concluded.

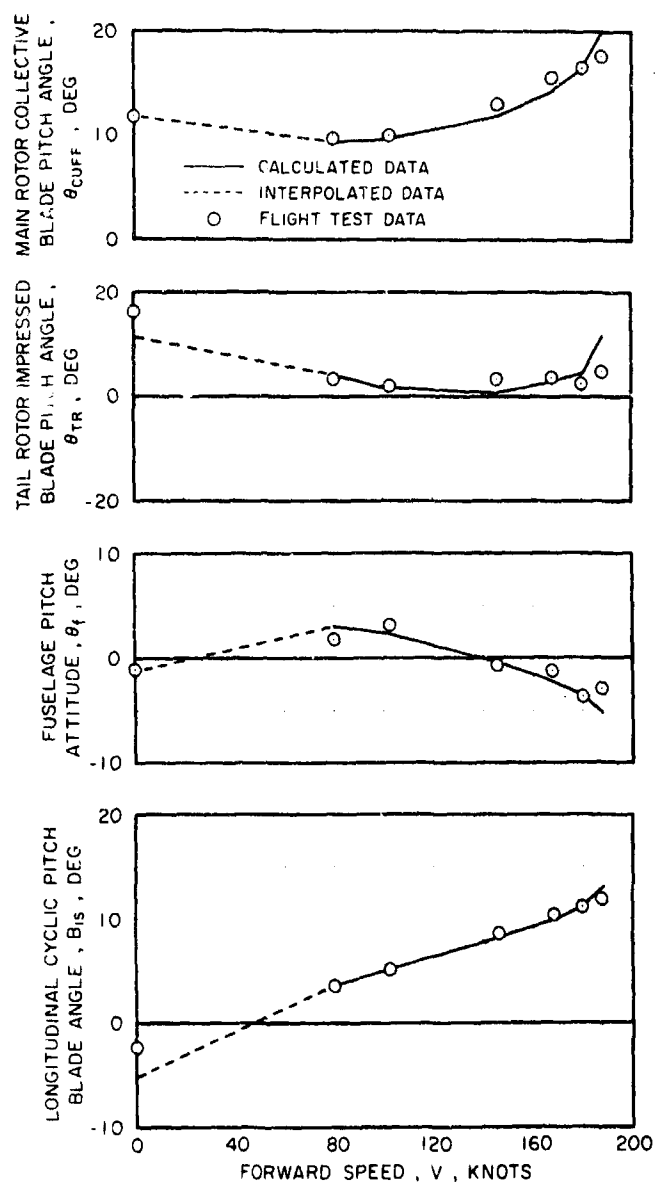93

(a) LONGITUDINAL TRIM PARAMETERS

Figure 39.   Correlation of Calculated Aircraft Trim
            Characteristics With Flight Test Data;
            Bias = 0 Deg, G.W. = 16,650 Lb, C.G. = 258 In.
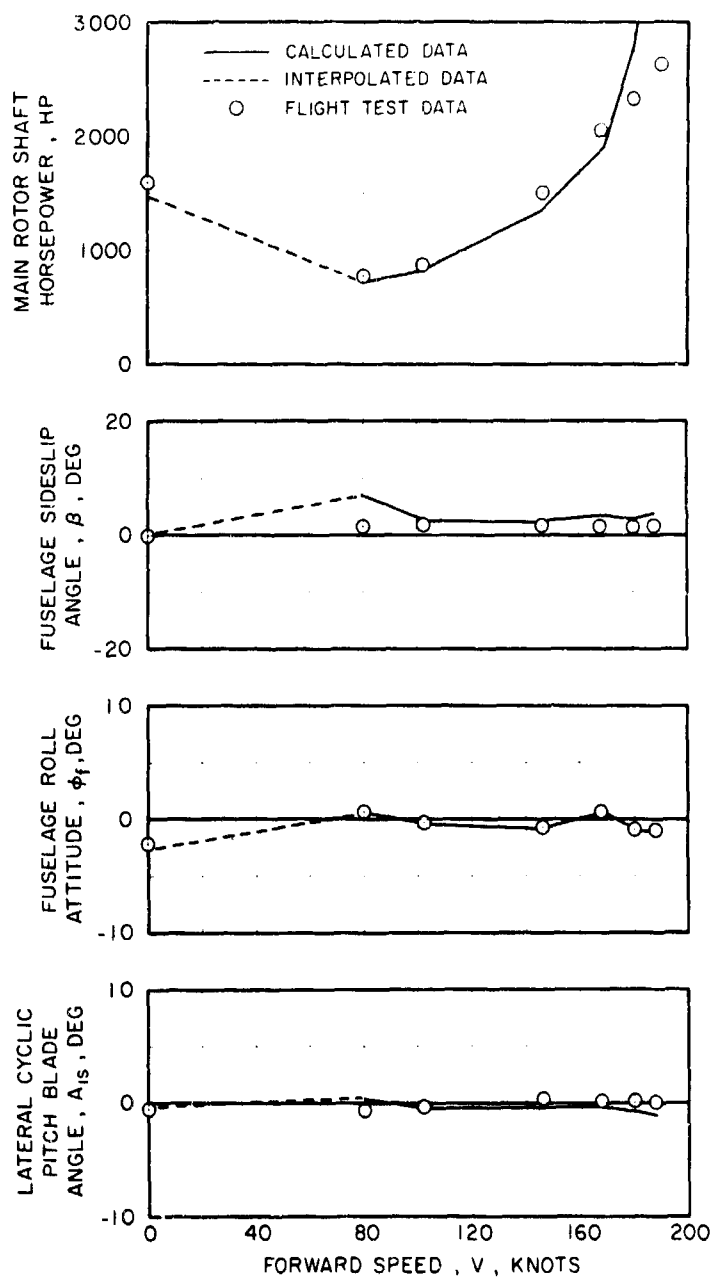
94

(b) LATERAL TRIM PARAMETERS

Figure 39. Concluded.

95

(a) LONGITUDINAL TRIM PARAMETERS

Figure 40.   Correlation of Calculated Aircraft Trim
             Characteristics With Flight Test Data;
             Bias = 0 Deg, G.W. = 16,800 Lb, C.G. = 258 In.

96

(b) LATERAL TRIM PARAMETERS

Figure 40. Concluded.